# 1995 Workshop on Trajectory Optimization Methods and Applications

Presentations from the Atmospheric Flight Mechanics Conference sponsored by the American Institute of Aeronautics and Astronautics, Baltimore, Maryland, August 7–10, 1995





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National Aeronautics and Space Administration

Office of Management

Scientific and Technical Information Program

1996

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#### **PREFACE**

#### 1995 Workshop on Trajectory Optimization Methods and Applications

The AIAA hosted the fourth "Workshop on Trajectory Optimization Methods and Applications" at the annual Atmospheric Flight Mechanics Conference on August 7, 1995 in Baltimore, Maryland. This workshop was last held at the 1993 conference in Monterey, California.

This workshop presented recent advancements in trajectory optimization programming and applications. The developers of trajectory optimization programs were invited to present a 15-20 minute summary of recent improvements or current work. Users of trajectory optimization programs, particularly those supporting flight programs, utilizing new approaches or solving unique problems, were also encouraged to present a short summary of their applications and/or modifications. This workshop provided a unique forum to exchange ideas and information between industry, academia, and government agencies involved in trajectory optimization.

This publication contains the charts presented at the workshop.

For more information please contact:

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#### 1995 Workshop on Trajectory Optimization Methods and Applications AIAA Atmospheric Flight Mechanics Conference

**Hyatt Regency** Ball Room B Baltimore, Maryland August 7, 1995 9:00-5:30

Chairs:

Steve Alexander

NASA Lewis Research Center

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#### 

\*\*Steve Paris

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Synopsis:

The newest version of OTIS, developed by the Boeing Company for NASA Lewis Research Center, is introduced. Version 3.0 improvements include: new sparse optimizer (SOCS), interplanetary capability, new phase input structure (allows easy phase addition-deletion), new equations of motion options (Lagrange planetary and spherical), non-state constraints at events, analytic phases, libration point and planet coordinate systems, planetary ephemerides, new attitude definitions (inertial "aerodynamic" angles, RTN, and inertial relative to the sun), calculation of shadow and planetary capture/escape parameters as well as Mode 3 improvements (fast derivatives and parallel shooting).

## 

- \*\*Koorosh Mirfakhraie, Frank Spurlock and Les Balkanyi
- \* Analex Corporation, NASA Lewis Research Center

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(fax) (216) 977-7125

Synopsis:

HITOP is a high thrust trajectory optimization code currently under development in the Advanced Space Analysis Office, in conjunction with ANALEX Corporation at the NASA Lewis Research Center. The code is written to primarily handle Expendable Launch Vehicles (ELV). HITOP incorporates two different optimization methods: Nonlinear Programming and Calculus of Variations (C of V). The user may use either optimization method alone, or may combine certain elements of both of them. The code allows the use of several different higher order variable step Runge-Kutta routines for integration. Work is ongoing to include a Two Point Boundary Value Problem solver called MUSN in order to solve the C of V problems for which the Euler-Lagrange equations, variation end condition equations, and intermediate constraint equations have been analytically derived.

Optimal Libration Point Targeting for the SOHO Mission Using OTIS  Mode 3 and Parallel Shooting
**Steve Alexander and Koorosh Mirfakhraie  * NASA Lewis Research Center email: steve@batman.lerc.nasa.gov telephone: (voice) (216) 977-7127
Synopsis:  A highly detailed launch vehicle model has been developed in support of the Solar and Heliospheric Observatory (SOHO) mission that will be launched in November, 1995. Using OTIS mode 3 (parameter optimization), parallel shooting and "fast" derivatives, the SOHO problem from lift off of the Atlas IIAS to spacecraft insertion at the Earth-Sun-Moon L1 libration point is optimized. The model includes over 115 independent variables, 70 constraints, and solar pressure as well as lunar and solar gravitational effects.
Optimization of Many Burn Orbital Transfers
**Greg Dukeman  * NASA Marshall Space Flight Center email: gdukeman@sail1.msfc.nasa.gov telephone: (voice) (205) 544-5464
Software Used for NEAR Heliocentric Trajectory Design
**David W. Dunham and James V. McAdams  * Johns Hopkins University Applied Physics Laboratory email: david_dunham@jhuapl.edu telephone: (voice) (301) 953-5609
Synopsis:  This talk describes the NEAR delta-VEGA trajectory to rendezvous with the asteroid 433 Eros, including a flyby of the interesting main-belt asteroid 253 Mathilde. The presentation discusses the NEAR mission design problem, describes how and why we use the SAIC Trajectory Optimizer, MAnE, SWINGBY, and NEAR_Sim, and discusses the auxiliary software we have written to manipulate some of the files to compute certain parameters and tables needed by operations personnel and science team members.

MAnE - An Experiment in Modernizing Mission Analysis Softwa	<u>re</u> 5
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\*\*Jerry L. Horsewood

\* AdaSoft, Inc.

email: horsewood@adasoft.com telephone: (voice) (301) 725-7014

(fax) (301) 725 0980

Synopsis:

MAnE is an acronym for Mission Analysis Environment, which is a system of mission analysis and trajectory optimization software tools that are integrated under the control of a graphical user interface. MAnE is used for the analysis and optimization of multiple leg, heliocentric missions which employ conventional, high thrust propulsion. It is a commercial product that is currently available for use on PC compatible computers.

The presentation focuses on the concepts which were the foundation for the design of the product and which distinguish it from other software that is currently available. The discussion closes with a brief summary of the components that are included in MAnE. Two papers, one providing a detailed overview of the features and capabilities of MAnE and one describing the use of MAnE in a case study, will be made available to workshop attendees.

### 

\*\* David Folta

\* NASA Goddard Space Flight Center email: Dave.Folta@ccmail.gsfc.nasa.gov telephone: (voice) (301) 286-6082 (fax) (301) 286-1763

Synopsis:

SWINGBY is an interactive program developed and supported by the Flight Dynamics Division at the Goddard Space Flight Center. It allows the user to accurately design and optimize trajectories, plan maneuvers, and analyze missions. While it was developed to support missions that use lunar gravity assists, libration point orbits, comet intercepts, and planetary missions, it can also be used to analyze low Earth orbits. Swingby was used to design and support the Clementine and Global Geospace Science (GGS) WIND missions and to design the SOHO and ACE trajectories.

SWINGBY is a DOS based program were the user interacts using pull down menus and displays of trajectories which can be presented in numerous coordinate systems, such as Solar Rotating, Libration Point, or Ground track Coordinates. SWINGBY displays graphical and numerical information as it is calculated to give the user real-time feedback. SWINGBY has several high fidelity numerical methods for targeting and propagating and can support finite burn maneuvers in any given spacecraft attitude.

NEAR-Sim: Near Earth Asteroid Rendezvous Simulation
**Greg Clubb and Victor Bond  * McDonnell Douglas Aerospace- Houston Division email: clubb@pat.mdc.com telephone: (voice) (713) 244-4433
Synopsis:  The Near Earth Asteroid Rendezvous Simulation (NEAR-Sim) was developed for the Applied Physics Laboratory (APL). NEAR-Sim provides precision trajectory targeting of APL's upcoming NEAR mission. Other simulation features include computation of an intermediate asteroid flyby, and computation of Deep Space Network (DSN) tracking information.
Development of a MATLAB Toolbox for Trajectory Optimization and Planning
**Mark L. Psiaki * Technion Israel Institute of Technology
email: mlp4@cornell.edu telephone: (voice) (607) 255-9100
Geometric Considerations in the Solution of Optimal Control Problems
Synopsis:  In the indirect approach to solving an optimal control problem, one is faced with solving a two-point boundary value problem for a Hamiltonian system of ordinary differential equations. The trajectories for the Hamiltonian system are often organized in a manifold structure dictated by stability and/or time-scale properties. Some ideas on how to design solution methods to take advantage of this structure are presented.

Recent Developments in Trajectory Optimization at McDonnell  Douglas
** Rocky Nelson  * McDonnell Douglas  email: nelson#m#_rocky@apt.mdc.com  telephone: (voice) (714) 896-1762  (fax) (714) 896-6930
Synopsis:  This talk presents current experience with trajectory optimization using OTIS and SNOPT, a large scale sparse SQP optimizer. In addition, our current experience modeling placards and tabular data is discussed. Applications are shown (HSCT and SSTO). Finally, the current use of a variational launch vehicle program, AF49, that was developed in the 60's is presented.
DAB Ascent Ready for WinNT/Win95
** Dab Engineering, Inc email: dbaker@csn.net telephone: (voice) (303) 757-6425
Some Recent Developments in Computational Optimal Control
** Dr. Renjith Kumar and Hans Seywald  * Analytical Mechanics email: rrk@phobos.larc.nasa.gov seywald@scb2.larc.nasa.gov telephone: (voice) (804) 864-1822
Synopsis:  Two new methods for the fast calculation of near-optimal trajectories are presented, namely a receding horizon non-linear feedback approach, and a concatenated trajectory optimization method for off-line optimization. Both methods show good performance and require only modest computing power.

Trajectory	Optimization f	or the High Speed Civil T	<u> Fransport</u>
* Honeyw			
email: sha telephone:		ywell.com (612) 951-7438 (612) 951-7438	rschultz@src.honeywell.com (612) 951-7257
	for different he portions of the over land segn supersonic ove state and calcu	orizontal paths between I route being over land are nents. The trajectories are er water. The optimal vert	ansport are discussed. Fuel usage is compared Dallas and London. Direct routes with large e compared to longer indirect route with shorter e constrained to be subsonic over land but can be ical plane trajectories are solved using the energy s. A method for solving the combined vertical
Trajectory	Optimization for	or a SAM Using a Multi-	Tier Approach 161
email: cph telephone: Synopsis:	urface Warfare urface Warfare illi%g20hp83: (voice) (fax)  This paper des method of opti simulations. T trajectory optii simulation to s the target. The models to prov The MTOP ap	5@relay.nswc.navy.mil (703) 663-4961 (703) 663-4166 cribes the Multi-Tier Optimizing the performance the method uses a multi-transport with a contraction software with a contraction software with a contraction of the method allows the use covide the Measure of Effective (703) 663-4961	timization Process (MTOP), which provides a of highly complex systems using high fidelity iter or multi-step approach to combine traditional complex Monte-Carlo terminal homing which truly maximize the probability of hit against of high fidelity 6 degree of freedom stochastic ctiveness (MOE) in the trajectory design process. blems where highly complex simulations are
Using OTI	S for Advance	d Conceptual Design	
email: leo telephone: Synopsis:	ney Associates nmck@maw.c (voice) (fax)  Results of app transportation	om (714) 536-0289 (714) 536-2317 lication of OTIS to devel	op concepts for an air-launched space with conventional ground-launched systems and em concept.

Application of	FONSIZE to Bi-prop and Tri-prop SSTO Designs		183
** Hai N Ngu			
* Aerospace C	Corp		
email: nguyer	@courier3.aero.org		
telephone:	(voice) (310) 336-4283		
•	(fax) (310) 336-5099		
Synopsis:		•	

FONSIZE is a trajectory/launch vehicle sizing program developed independently at The Aerospace Corporation. The program allows optimization of the trajectory and vehicle sizing simultaneously, thus, providing accurate results for highly sensitive designs such as Single Stage to Orbit vehicles. The weights of the vehicle are represented by a set of nonlinear algebraic equations containing design and sizing variables which are iterated during the optimization process. The trajectory is divided into segments and each segment is approximated by a third order polynomial. Coefficients of the polynomials and control profiles are iterated to optimize the objective function subject to constraints such as sizing, design, and trajectory. The program has been used extensively to assess and evaluate new launch concepts, both expendables and reusables.

#### 

- \*\* Diane Buell
- \* MITRE Corporation email: ddbuell@mitre.org

telephone:

(voice) (703) 883-3738 (703) 883-5963 (fax)

Synopsis:

The Earth Satellite Program is a user-friendly, highly graphical, Macintosh-based program that was developed for the Air Force Milstar program to support satellite constellation and ground terminal design studies. The computer program generates satellite ground tracks, determines outage zones for a constellation of satellites, and generates spot beam projections.

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## **Overview of OTIS 3.0**

Presented by

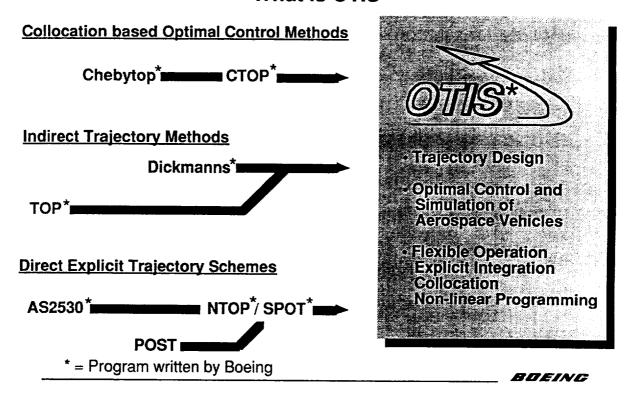
Steve Paris Boeing Defense & Space Group swp@milo.ds.boeing.com

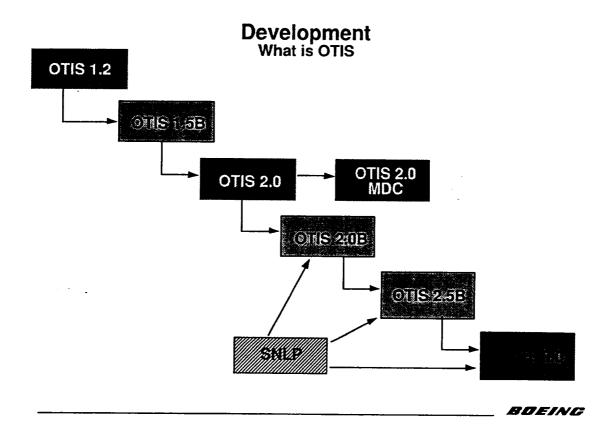
BOEING

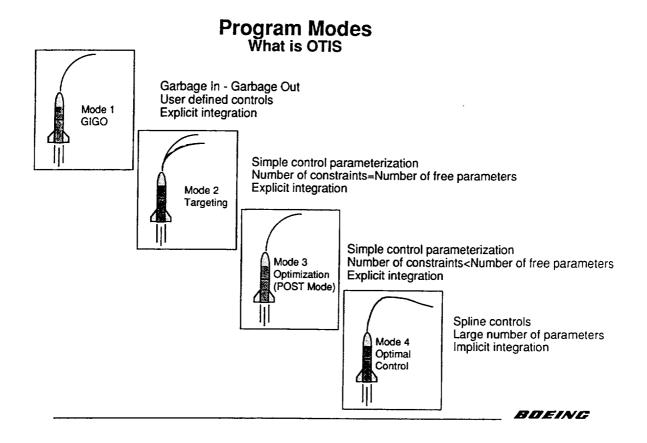
## **Agenda**

What is OTIS
Why OTIS 3.0
OTIS 3.0 Features

#### What is OTIS



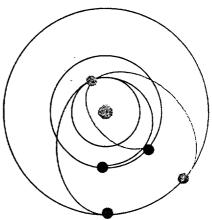




### New Solutions Why OTIS 3.0

## **Old Trajectories**

## **New Trajectories**



Ballistic Paths Low Thrust Escapes Planetary Flybys Aerobraking Free returns

## Desired OTIS Features Why OTIS 3.0



**Reduce Memory Requirements** 

## **Reduce CPU Consumption**

**Provide Better Interface** 

400<sup>+</sup> inputs + 600 <sup>+</sup>outputs = User frustration and errors



**Provide More Simulation Flexibility** 



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## **OTIS 3.0 Objectives**

#### Provide more Flexible simulation

- Coordinate systems
- Phase structure
- Central Body shifts
- Different equations of motion Lagrange Planetary Equations Spherical Coordinates
- Ephemerides

#### **Faster Solutions**

- Sparse NLP
- Different Quadrature formulas

#### Start from OTIS 2.5B

#### **OTIS 3.0 Implementation**

#### Simulation (All Modes)

- Number of controls per phase can change
- Different control type per phase
- Number of state variables can change per phase

#### Explicit Modes (1,2 &3)

- Adams Integrator
- Fast Derivatives
- Parallel Shooting
- Restart Files

#### Mode 4

- NPSOL Memory Management (Poor mans sparse)
- SOCS Interface / Node Refinements/Defect Formulas
- Analytical Jumps
- Global Event Constraints
- Quintic Defects

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## New Phase Philosophy

**OTIS 3.0 Implementation** 

**OTIS 2.0** 

**OTIS 3.0** 

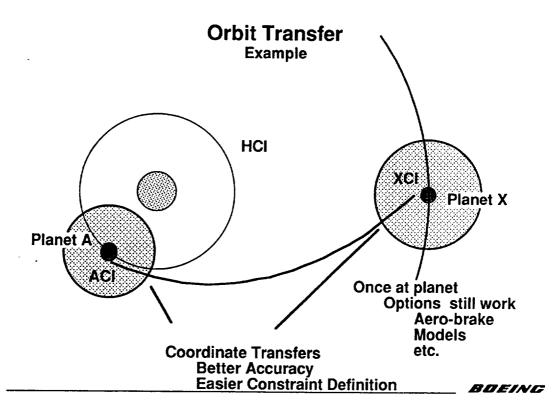
Phases linked directly (straight mapping) x\_=M[x<sub>+</sub>]

Phases linked as constraints  $x_- \Phi[x_+]=0$  added as "defects" for phases P2 and P4

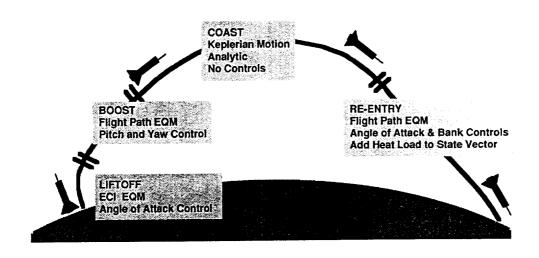
## Why is this better? OTIS 3.0 Implementation

## More flexibility

- Modular structure
- Allows for mixing of phase types
  - Collocation
  - Analytical
  - Transformation
    - » Equations of Motion
    - » Central body
- · All Phases look the same
  - Input
  - Print out
  - Trajectory generation



#### ICBM Example



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#### **OTIS 3.0 Results**

#### Faster and more compact

#### Supersonic Interceptor Minimum Time Climb (Bryson Problem) Working Array Size OTIS - OTI ODE error **CPU Time** Number OTIS-SOCS OTIS -OTIS-SOCS of nodes NPSOL **NPSOL** 0.27 28492 67883 29.44 13.53 10 0.057 86215 44.79 107172 59.69 20 105408 0.034 178.98 75.21 236252 30 124944 0.012 415732 124.8 40 558.3 3.00E-07 177428 224.47 57

#### **NASA LeRC's Involvement**

## Testing of Integrated Program

- Verify OTIS3 satisfies defined requirements
- Match OTIS3 results against known variational solutions (where possible)

#### Documentation

- Vol. IV Applications Manual
- Testing results

#### Distribution

- SOCS only available from BCS with site license
- OTIS3 source is public domain
- LeRC likely to assume responsibility for publication and dissemenation of documentation
- Web Site to handle "OTIS3 subscription"

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### **OTIS 3.0 Developement Schedule**

## Key Events

- Delivery of OTIS3 for testing in August including updated SOCS
- Completion of Testing in October
- Completion of Documentation in October
- OTIS3 Training at LeRC in late October
- Completion date of all OTIS3 related activities in December

## **Purgamentum Init Purgamentum Exit**

welcome to the next level

OTIS 3.0 provides another step in the state of the art Quality has increased with a decrease in cost

## HITOP: A High Thrust Trajectory Optimization Code

O.F. Spurlock \*

L.R. Balkanyi \*

Presented By

K. Mirfakhraie \*\*

Αt

Trajectory Optimization Methods and Applications Workshop

AIAA Atmospheric Flight Mechanics Conference

August 7, 1995

- Advanced Space Analysis Office NASA Lewis Research Center
- • Analex Corporation

ANALEX Corporation

NASA Lewis Research Center Advanced Space Analysis Office

#### Outline

- Introduction
- Objectives
- Methods
- Present Status
- Future Work

#### Introduction

- Trajectory optimization of Expendable Launch Vehicles (ELV's) at the Advanced Space Analysis Office (ASAO) at LeRC is performed for:
  - Mission design for approved programs
  - Feasibility and planning studies
  - Corroboration of contractors' data for NASA missions flown on Atlas and Titan
- The computer program DUKSUP was written at LeRC during 1960's and early 70's as LeRC's primary tool for trajectory optimization.

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## Introduction (Cont'd)

- DUKSUP is a 3-D.O.F. code written for performance analysis of multi-stage high-thrust launch vehicles.
- Calculus of Variations (COV) is used to formulate the problem. The resulting two-point boundary value problem (TPBVP) is solved using a Newton-Raphson algorithm.
- Missions simulated with DUKSUP include:
  - Launches from ER and WR
  - LEO, GTO, and GSO insertions
  - Interplanetary escape trajectories
  - Orbit transfers

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## Introduction (Cont'd)

- Motivation in replacing DUKSUP:
  - Difficulty in modifying and expanding the code due to lack of documentation and outdated programming practices
  - Sensitivity to initial guesses
  - Difficulty in reformulating the COV problem when adding new features and constraints to the code
  - New advances in the field of trajectory optimization

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## **Objectives**

- Develop a fully documented and structured code
- In addition to retaining and enhancing the COV capabilities, employ full parameter optimization capabilities such that, depending on the problem, the following three methods may be used in solving trajectory optimization problems:
  - COV alone
  - Parameter optimization alone (with optional variational optimal steering)
  - A combination of COV and parameter optimization

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## Objectives (Cont'd)

 Develop pre- and post-processors to provide the user with maximum flexibility in modeling various ELV's and missions and manipulating and utilizing the data from the resulting optimal trajectories

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#### **Methods**

- COV is an indirect method of optimization. The
  necessary conditions for an optimal solution are derived
  variationally leading to a set of differential equations and
  corresponding boundary conditions, i.e. a two-point
  boundary value problem (TPBVP).
- Solution of the TPBVP will yield the optimal solution.
- Robustness of COV method depends on the method used to solve the TPBVP.
- Assuming 'good guesses', very rapid convergence is achieved. In DUKSUP large case studies, e.g. launch window performance studies, require relatively short CPU times.

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## Methods (Cont'd)

- In COV each problem requires developing its own set of variational equations (and subsequent coding).
- Parameter optimization is a direct method of optimization in which the problem is formulated in terms of a non-linear programming (NLP) problem in which the controls (independent variables) have to optimize the objective function subject to a set of constraints imposed on the problem.
- In general, a numerical algorithm is employed to solve the resulting NLP problem.
- Parameter optimization is more numerically intensive, and will have an impact on the robustness and CPU time requirements.

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### Methods (Cont'd)

- Formulating new problems with parameter optimization is much easier and requires minimum changes to the code.
- Combining COV and parameter optimization methods in solving a given problem will draw from the strengths of the two methods.

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#### **Present Status**

- Currently one civil servant (NASA) and four support service contractors (Analex) are developing HITOP.
- COV method:
  - Problem formulation and derivation of variational equations are almost complete.
  - Evaluation of TPBVP solver is complete. COLSYS (based on collocation) and MUSN (based on multiple shooting), both available from NETLIB, have been evaluated.
  - MUSN has been selected as the routine that will be used to solve the TPBVP arising from the COV approach.

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### Present Status (Cont'd)

- COV code, including MUSN, is being put together and tested concurrently.
- Parameter optimization method:
  - NPSOL has been selected to solve the resulting nonlinear programming (NLP) problem.
  - A Runge-Kutta algorithm based on the Fehlberg method has been selected as the integrator. It has both fixed and variable integration step size options.
  - The parameter optimization part of the code is undergoing test.

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## Present Status (Cont'd)

- Preliminary test cases on an Atlas vehicle model for a GTO mission have successfully been run.
- The independent variables (controls) for these tests consist of:
  - i) initial pitch and yaw angles and their initial time rates,
  - ii) variable burn and coast times,
  - iii) 'kick' angle, and
  - iv) discontinuity (jump) factors for Lagrange multipliers (to accommodate intermediate constraints).

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## Present Status (Cont'd)

- The code can handle a wide range of intermediate and final constraints. For the test cases, the intermediate constraint is the radius of perigee and the final constraints are the radius of perigee, radius of apogee, inclination, and argument of pericenter.
- Tests so far demonstrate the ease of adding and changing variables and constraints in this method.
- Producing 'good' guesses and values for various NLP solver parameters seem to be essential in obtaining a solution.
- CPU time requirements have been much greater than the runs with COV-based DUKSUP.

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## Present Status (Cont'd)

- COV/Parameter Optimization method:
  - The code development is underway.

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#### **Future Work**

- Alpha version of HITOP (with limited capabilities) is due out at the end of August '95.
- More testing and code development will lead to the beta version of the code to be released in December of '95.
- The operational version of the code will be released to the ASAO users in April '96. It will have full modeling capabilities for Titan II, III, and IV; and Atlas II, IIA, and IIAS vehicles and any other general launch vehicle with unique modeling requirements.
- Other launch vehicle and mission modeling enhancements will be made to the code in the future as required.

ANALEX Corporation

## Optimal Libration Point Targeting for the SOHO Mission using OTIS Mode 3 and Parallel Shooting

#### Steve Alexander Koorosh Mirfakhraie Mike Carney

Advanced Space Analysis Office NASA Lewis Research Center

Presented at:
Workshop on Trajectory Optimization Methods and Applications
AIAA Atmospheric Flight Mechanics Conference
Baltimore, Maryland

August 7, 1995

**Advanced Space Analysis Office** 

## **Background**

- Advanced Space Analysis Office (ASAO) at the Lewis Research Center (LeRC)
  performs mission analysis for NASA's Atlas and Titan Expendable Launch Vehicle
  (ELV) missions.
- LeRC has performed Atlas and Titan Mission design mission design since 1962.
  - Examples: Surveyor, Lunar Orbiter, Mariner, Pioneer, Helios, Viking, Voyager, Viking, Mars Observer, GOES
- A Calculus of Variations (C of V) code, written in the 1960's, is used to optimize these trajectories.
  - Solves two body problem
  - - Utilizes detailed launch vehicle models(thrust characteristics, weight flows, etc)
  - Targets to "near Earth" conditions (i.e., spacecraft/launch vehicle separation conditions)

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## **Background**

- In January, 1994 ASAO began developing a tool to augment the current C of V capability to support the Solar and Heliospheric Observatory (SOHO mission).
- This new capability would:
  - Optimize the complete problem from lift-off to insertion at spacecraft final orbit at the Earth-Sun-Moon L1 Libration Point (L1 or Libration Point Targeting)
    - C of V code can only optimize problem from lift-off to near Earth targets
  - Utilize detailed launch vehicle models (thrust characteristics, weight flows, etc)
  - Accurately model a 4 X 4 Earth gravity field, lunar and solar gravity forces, and solar radiation pressure during transfer orbit
  - Support SOHO mission design process (November, 1995 launch)
  - Support future missions (Cassini: October, 1997 launch)

**Advanced Space Analysis Office** 

### SOHO Overview

- SOHO is an international project that is part of the International Solar and Terrestrial Physics (ISTP) program
  - Program is jointly run by NASA and the European Space Agency
  - SOHO will investigate the Sun's corona, solar wind, and solar irradiance

Spacecraft builder: European Space Agency / Matra

Launch Vehicle: Lockheed Martin Atlas IIAS

Current Launch Date: November 7, 1995

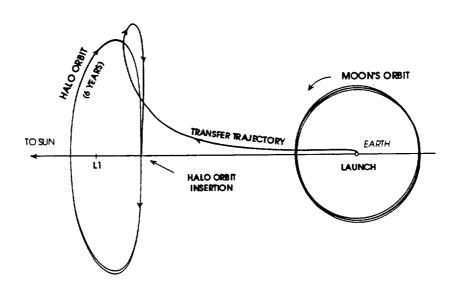
Transfer Time to Final Orbit: 3 to 4 months

Final Orbit: Halo Orbit about the Earth-Moon-Sun

L1 Libration Point (~ 1.5 million km from Earth along Earth-Sun line)

Life Time: 2-6 years

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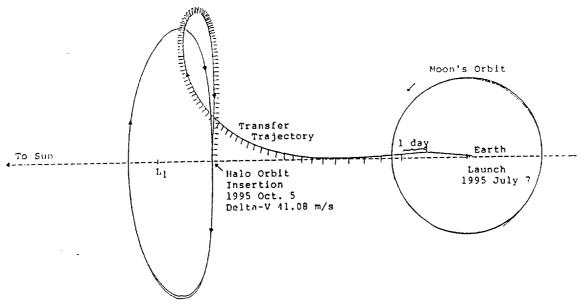


Direct Transfer to a Quasi-Periodic Halo Orbit View: Above Ecliptic Plane

Source: NASA GSFC

#### **Advanced Space Analysis Office**

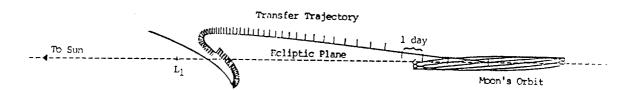
## SOHO Mission Overview SOHO Transfer Orbit



Direct Transfer to a Quasi-Periodic Halo Orbit View: Above Ecliptic Plane

Source: NASA GSFC

#### SOHO Mission Overview SOHO Transfer Orbit

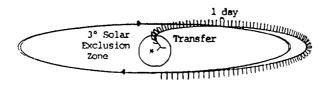


Direct Transfer to a Quasi-Periodic Halo Orbit View: In Ecliptic Plane, Side View

Source: NASA GSFC

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## SOHO Mission Overview SOHO Transfer Orbit



Direct Transfer to a Quasi-Periodic Halo Orbit View: In Ecliptic Plane, Looking toward Sun

Source: NASA GSEC

### **Tool Selection**

- A specially modified version of OTIS (Optimal Trajectories by Implicit Simulation), using the parameter optimization mode (Mode 3), was chosen as the new optimization tool.
  - LeRC in house C of V code not a viable option since it is not capable of solving the Nbody problem required for L1 targeting.
  - A moderate amount of OTIS experience existed in house since 1987.
  - The collocation mode in OTIS, without access to a sparse optimizer, could not model the problem accurately enough to support the SOHO mission.
  - ASAO had successfully used Mode 3 to support the Advanced Communications Technology Satellite (ACTS) flight operations for STS-51 in September, 1993.
  - OTIS could easily be modified to accept a LeRC Nbody propagator to model
    - LeRC Nbody propagator already had been qualified to support SOHO mission

**Advanced Space Analysis Office** 

### New OTIS Features used for SOHO Mission Design

- Fast Derivatives
  - Improves the efficiency of calculating the Jacobian by integrating only the portion of the problem that is affected by the perturbation of the particular independent variable.

Reduced time to calculate Jacobian by an order of magnitude

- Phase Dependent equations of motion and number of controls
  - The type of equations of motion and the number of controls can vary between phases allowing greater flexibility in defining the problem.

At lift-off, the robust cartesian equations of motion are employed to avoid singularities. Later phases employ more efficient, but less stable, equations of motion.

- · Parallel Shooting or Multiple Shooting
  - Trajectory initially broken into separate segments to improve optimization.

Greatly improved the robustness of the SOHO problem

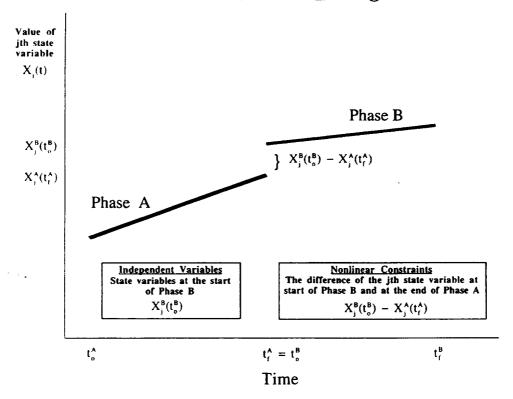
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### **Parallel Shooting**

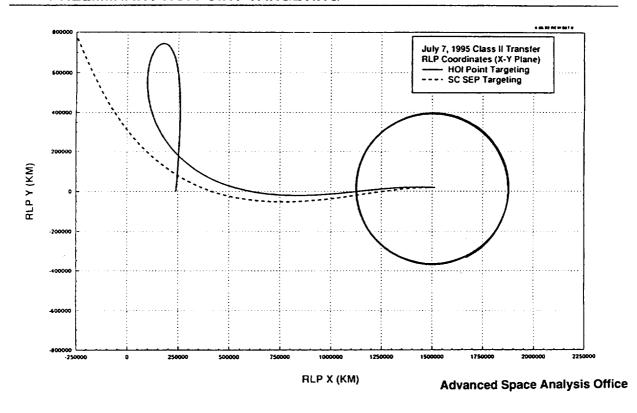
- Parallel Shooting Nodes (PSN) are designated only at the end points of phases
- Each PSN creates 7 additional independent variables and 7 additional equality constraints (one for each state variable)
  - Continuity for each state variable is enforced at a PSN by specifying the difference of each state variable at the node as nonlinear equality constraints
  - These constraints are satisfied (continuity enforced) at the end of the optimization
- Parallel Shooting in Mode 3 is similar to collocation in OTIS (Mode 4) except:
  - PSN are only at the end points of a phase
  - In Parallel Shooting the states are determined by the actual equations of motions instead of cubic approximations as in Mode 4
    - OTIS collocation segments must be short to accurately represent the states using cubic approximations/implicit integration
- The advantages of Parallel Shooting are:
  - Nonlinear, highly sensitive problems are made more linear and less sensitive
  - Increased robustness
  - · Greater control over the initial trajectory
  - · Initially breaks the problem into separate pieces

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### Parallel Shooting

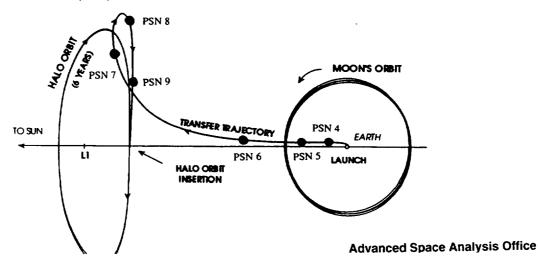


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### Parallel Shooting in the SOHO Problem

- Nine Parallel Shooting Nodes are used in the SOHO problem
  - One during powered ascent
  - One after spacecraft injection
  - One at spacecraft separation
  - Six after spacecraft separation (Nbody coast phase)
- Using default tolerances, the inaccuracy due to Parallel Shooting discontinuities is less than 25 km out of 1,500,000 km



### **Summary of SOHO Solution**

• Number of Parallel Shooting Nodes: 9

• Number of Independent Variables: 115

PSN Independent Variables

• Explicitly Declared 52 Burn times, pitch/yaw steering coefficients, etc

63

• Number of Dependent Variables/Constraints: 73

• PSN Constraints 63

• Explicitly Declared 10 Park orbit targets, powered ascent constraints, L1 targets

• Computing Platforms:

• Primary Sun SPARCserver 1000

• Secondary Cray YMP

• Time to compute Jacobian: ~ 400-500 sec (Sun)

• Optimizer: NPSOL 2.1

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### **Conclusions**

- OTIS Mode 3 with Libration Point Targeting is successfully supporting the SOHO flight program
  - · Development and check out are complete
  - · Launch windows are currently being run for every launch day
  - Mode 3 and NPSOL can support "large" number number of variables
- Parallel Shooting is a powerful optimization method
  - Greatly improved convergence for SOHO mission
  - Nonlinear, highly sensitive SOHO problem is made more linear and less sensitive

### Optimization of Many-Burn Orbital Transfers\*

### August 7, 1995

### AIAA Guidance, Navigation, and Control Conference, Baltimore, MD

John M. Hanson and Gregory A. Dukeman\*\*

\* This research supported by the MSFC Center Director's Discretionary Fund, project 94-15.

\*\* Aerospace Engineers, Flight Mechanics Branch, NASA Marshall Space Flight Center, MSFC, AL 35812

### Overview

- Introduction
- Background
- Approach
- Individual Burns
- · Parametric Optimization of Many Burns
- Test Cases
- Summary

### Introduction

- motivation: develop efficient algorithms for optimizing low and medium thrust orbital transfers that use a significant number of burns
- acceleration levels too high to use orbit averaging and too low to use only a few burns
- combine direct optimization methods with indirect methods to reduce the sensitivity of the orbital transfer problem and reduce the number of parameters to optimize
- (mathematically) sub-optimal solutions generated but very close to published optimal results
- examine transfers to: 1) HEO, 2) GEO, 3) GPS orbit, 4) lunar SOI, 5) escape orbits

### Background

- OTIS program (1990) used collocation techniques to optimize 3-burn escapes from Earth orbit
- Kluever and Pierson (1994) use up to 5 burns in transferring from a medium Earth orbit to lunar SOI
- Redding and Breakwell (1984) developed a COV solution for transfer from LEO to GEO with up to 16 burns
- solutions for AXAF 19-burn transfer found using suboptimal methods
- Betts (1993) generated continuous low-thrust solutions with sparse nonlinear programming/direct transcription
- Breakwell and Chanal (1986) examine many-burn transfers where acceleration is low enough to use orbit averaging; ignored oblateness effects

### Problem Formulation

- max thrust/coast/max thrust/... transfers only
- · constant Isp and mass rate, specified initial mass
- · all cases begin with a number of perigee burns
- for geocentric orbital transfers, apogee burns establish the final orbit conditions
- for transfer to escape, final perigee burn provides desired excess energy
- for lunar case, final perigee burn takes spacecraft
   to SOI and final burn there gives desired conditions

### Problem Formulation (cont'd)

- · orbit transfer optimization problem:
  - determine thrust direction history and engine on/off times that minimize total fuel usage for a fixed number of burns and satisfy endpoint trajectory constraints
- tradeoffs can be made between number of burns and total transfer time; as number of burns increases, typically fuel usage goes down but trip time increases

### Two Extreme Solution Methods

- Indirect Method
  - use optimal control theory to get MPBVP involving switching conditions and end-point conditions
  - Advantages: 1) Small number of unknowns, i.e., switching times and initial costate,
  - 2) numerically very efficient
  - · Disadvantage: difficult to actually get a solution
- Direct method:
  - parameterize the thrust direction history (in some suitable form) to obtain a constrained parameter optimization problem
  - Advantage: simpler formulation
  - Disadvantage: many parameters to determine

### **Current Solution Method**

- 1) use modified parameter optimization to determine the finite set of parameters, i.e., engine on/off times and trajectory targets for each individual burn and
- 2) use optimal control theory to obtain the infinite set of parameters, i.e., thrust direction histories for individual burns

Advantage: reduced sensitivity

Modified Parametric Optimization

- need to determine start and stop times of many burns along with targets (e.g., perigee, apogee, inclination) for each burn
- large number of burns ---> large number of parameters
- let a particular type of parameter (say perigee targets) be represented by a polynomial in burn number n:

parameter type  $=a_0^{}+a_1^{}n+a_2^{}n^2$ so that 3 polynomial coefficients need be optimized instead of several perigee targets

### Why Quadratic Polynomials?

- burn duration vs burn number data is fairly smooth
- quadratic form because higher order forms don't increase performance enough to justify the extra parameters
- past work shows that very low thrust transfers are insensitive
- comparison of results herein with published results validates quadratic form

### Parameter Optimization Method

- brute force unconstrained optimization algorithm
  - varying one parameter at a time, fit a quadratic and estimate the peak of the curve (max final mass)
  - this method allows for rigid control of increments taken during optimization

### Individual Burn Optimization

- modified version of the OP GUID guidance algorithm described in NASA CR-1430:
  - numerically solves the associated TPBVP from optimal control theory for a single burn
  - closed-loop/open-loop
  - Earth oblateness/inverse square
  - more than 40 sets of mission objectives available involving min/maximization/fixing of endpoint quantities, e.g., apogee, perigee, semi-major axis, inclination, final time, etc.

### OP GUID (cont'd)

- burn objective examples
  - minimize burn duration subject to prescribed values for final semi-major axis, eccentricity, and inclination
  - maximize apogee subject to prescribed values for final perigee, inclination, and burn duration
- single burn optimization is very quick, routine and mostly automated; ad hoc checks for non-optimal local extrema; burn times longer than an orbit difficult

### **End-Point Necessary Conditions**

- much of the modification to OP GUID involved deriving constraints on final costate for the new mission objectives
  - •Maple symbolic manipulation software was used to efficiently derive the required expressions and auto-generate FORTRAN code
  - •paradigm of analytic partial derivatives in OP GUID further increased the utility of Maple
  - •symbolic software decreases development time from months to hours

### Test Cases

- 1) Redding's 5-burn transfer to geostationary orbit
- 2) OTIS 3-burn escape from orbit
- 3) Kluever's 5-burn transfer to the lunar sphere of influence
- 4) 3-burn escape with lower acceleration than case 2
- 5) 18-burn transfer to GEO
- 6) AXAF 19-burn transfer to HEO
- 7) Hundreds-of-burns transfer to geostationary orbit with low thrust

# Test Cases (cont'd) 5-burn transfer to GEO

initial orbit: 6600 km circ, incl = 28.5 deg final orbit: 42241 km circ, incl = 0 deg no Earth oblateness, Isp = 450 s, m0 = 4231.2 kg, thrust = 2000 N, (0.05 g's)

final mass: Redding: 1592 kg,

current method: 1593.7kg

### Test Cases (cont'd)

OTIS 3-burn escape from orbit

initial orbit: 700 km alt circ final orbit:  $C3 = 10.48 \text{ km}^2/\text{s}^2$ 

includes Earth oblateness, Isp = 950 s

m0 = 372000 kg, thrust = 667,233N, (0.18g's)

final mass: OTIS: 252000kg

current method: 252273kg

number of parameters used: 7

### Test Cases (cont'd)

Kluever's 5-burn transfer to lunar SOI

initial orbit: 13621.77 x 11729.65km

final orbit:  $41497 \times 36994 \text{ km}$ no Earth oblateness, Isp = 10046 s

m0 = 96862kg, thrust = 2942N, (0.003g's)

final mass: Kluever: 94627kg

current method: 94749kg

### Test Cases (cont'd)

3-burn escape with lower acceleration than case 2

initial orbit: 700km alt circ final orbit: C3=14 km $^2/s^2$ Earth oblateness, Isp = 925 s m0 = 453592kg, thrust = 133446N, (0.03g's)

final mass: current method: 273397kg

number of parameters used: 4

### Test Cases (cont'd)

### 18-burn to GEO

initial orbit: 185km alt circ, incl=28.5 deg final orbit: 35786km alt circ, incl = 0 deg Earth oblateness, Isp = 311 s m0 = 20982kg, thrust = 1846N, (0.009g's)

final mass: current method: 5041kg

### Test Cases (cont'd)

### AXAF 19-burn transfer to HEO

initial orbit: 250km alt circ final orbit: 10000km x 100000km Earth oblateness, Isp = 311 s m0 = 20982kg, thrust = 1846N, (0.009g's)

final mass: current method: 7087kg

number of parameters used: 8

### Test Cases (cont'd)

### Hundreds-of-burns transfer to GEO

377 perigee burns, 100 apogee burns initial orbit: 185km alt circ, incl=28.5 deg final orbit: 35786 km alt circ, incl=0 deg Earth oblateness, Isp = 860 s m0 = 12247kg, thrust = 40N, (0.0003g's)

final mass: current method: 6492.7kg

### Summary

- In all cases where literature gave results suitable for comparison, the present method equalled or exceeded those results
- have shown that current methods applicable to many types of useful transfers, thrust levels and numbers of burns
- in general, the degree of optimality of these methods is unknown although the numerical testing suggests it is very close in many useful cases

### Summary (cont'd)

- Possible extensions:
  - -- use more efficient parameter optimizer (e.g., NPSOL)
    - -- make improvements to OP GUID to improve its convergence properties (e.g., multiple shooting)

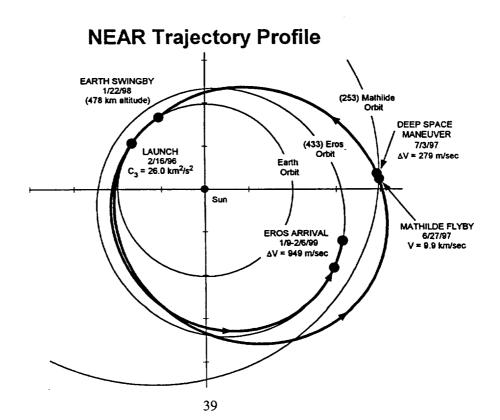
### AIAA Trajectory Optimization Workshop August 7, 1995

# Software used for NEAR Heliocentric Trajectory Design

David W. Dunham and James V. McAdams

Phone: (301) 953-5609 Fax: (301) 953-6556

E-Mail: david\_dunham@jhuapl.edu



### Major Software Packages used for NEAR

**SAIC Trajectory Optimizer** 

Mission Analysis Environment (MAnE) - Adasoft

Swingby - Goddard S.F.C. & Computer Sci. Corp.

NEAR\_Sim - McDonnell-Douglas Aerospace, Houston

### Other NEAR Software

Interface Programs - Mulpoint, Injstate (FORTRAN)
Utility Programs - Mindist, Angldist, Flybyswg
Compilers - Lahey F7713, Graphoria (CalComp plot compatible)
Stanford Graphics - for good-quality plots
PCWrite - for editing text files

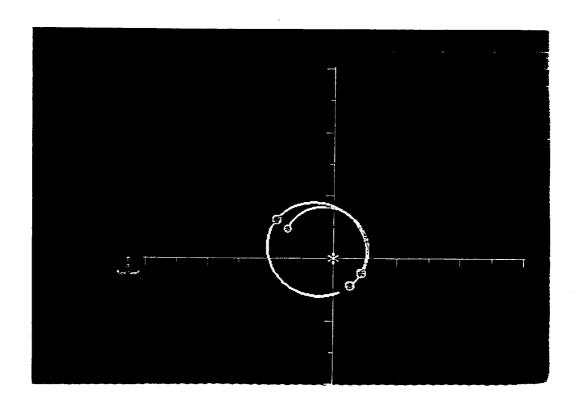
### **SAIC Trajectory Optimizer**

**PC-based** 

Used for first NEAR trajectory designs

Heliocentric Keplerian zero sphere-of-influence

Easy convergence of complex problems



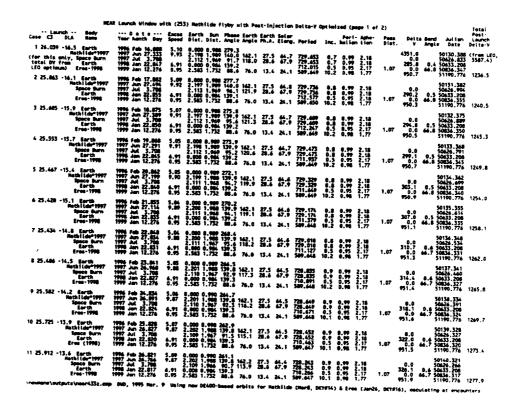
# Mission Analysis Environment (MAnE)

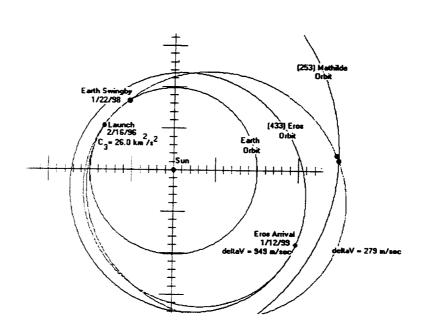
PC-based - Windows

Heliocentric Keplerian zero sphere-of-influence

Great user interface, easy to use

Variety of output options, including labelled plots





### **Swingby**

PC-based (DOS)

Precision full force-model for near-Earth and heliocentric phases

Good menu-driven user interface

Launch model - targeting by varying launch time, coast time, and injection velocity

Easy coordinate transformations

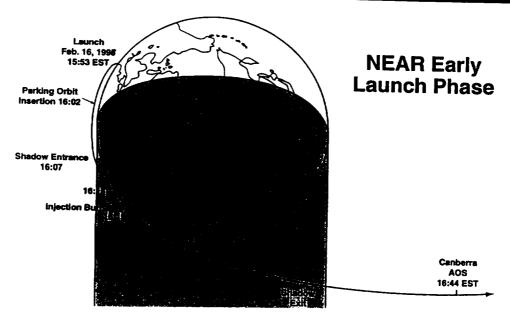
Displays trajectories in many different ways, useful for plots for presentations

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# Near Earth Asteroid Rendezvous



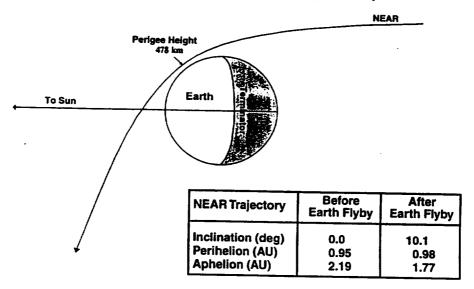




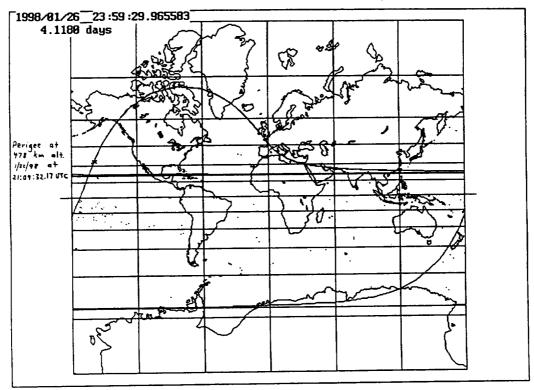
# Near Earth Asteroid Rendezvous



# Close-up of Earth Flyby, Jan. 22, 1998, Side View



# Ground Track During Earth Swingby, Jan. 22, 1998, With Nominal Launch, Feb. 16, 1996



### NEAR\_Sim

Based on modern ASDS software, written in Ada for workstations

Precision full force-model for near-Earth and heliocentric phases

Overlap method of targeting  $\Delta VEGA$  trajectories for rapid and robust convergence

Extensive ephemeris tables, can be used for plots

Detailed station coverage information

	NEAR_Sim	9995		•			(km/s)	
1.0000	-1.827126	503.354141993668		*00 (km/s)		00 (km/s) 00 (km/s)		iO (km/s) 1 (km/s) m/s)
0.000000073			E	Time: 1996/02/16 21:18:39.76 (utc_uano)  Mass: 805.000 (kg)  Earth J2000 Equatorial Position & Velocity -1902.66502150211 6257.73328281335 -501.503454392987 (km) -100.0834412159008 -3.53595231523604E+00 -5.7713029272412E+00	9.9338	286918277,77783 71652184, 9657164 522360,06602133 (tw.) -6.334459627660146+00 15.751118414622 6.675211696364456+00 Asserted J2000 Gemetoriel Posticion & Velocity 8.74826067240786e-01 22.5844434561340 6.78845904327206+00	-7.2028568920093E+00 -6.83335159465179E+00 -1.13243346907850E-01	lime: 1997/07/03 17:00:00.00 (utc_usno)  leas:  605.0000000000 (kg)  You 22000 Equatorial Desition & Velocity 23:199987.17344. 263:1062.5347773 8894370.27643260 (km)  A.1663046998385-00 15.6869896201600 6.657245802564:156-00  You 72000 Ecliptic Position & Velocity 23:199987.17345. 22644115.9383311 - 236466.37465563 (km)  A.16630469958385-00 17.0416712958661 - 1.3465302802039;E-01  arth 42000 Equatorial Position & Velocity 262020000.231245 162995466.213729 66115423.5697528 (km) 35:6491842708433 10.1882627727194 4.2703783193594228-00 (km/.
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### Integrated Results for NEAR, Opening Day of Launch Period Final Results (3/27/95)

												,,-	-,		
Launch C3	Event Name	;	Date Month	 Day	G.M.T. HHMMSS	Excess Speed*	Earth#	Sun# Dist.	Phase@ Angle	Earth Angle	Solare Elong.	Pass Dist.	Delta	t-Injection Delta-V/ Total Delta-V	Poweree Pactor
25.985	Launch(TTI) Hat. Flyby DSM 1 DSM 2 Earth SWB Rend. 1 Rend. 2 Rend. 3 Rend. 4 Eros Flyby	1996 1997 1997 1998 1998 1999 1999 1999	Feb Jun Jul Jul Jan Jan Jan Jan Jan Jen Jan Jen Jen Jen Jen	3 4 22 9 16 23 30	211839 075752 170000 170000 210932 170000 170000 170000 170000	5098 9934 6900 250 50	182km 2.198 2.112 2.098 478km 2.574 2.595 2.611 2.622	0.988 1.989 1.968 1.965 0.984	140.0 77.8 77.7 139.2 84.4	117.3 117.7 105.4 101.6 97.7	64.699	1.028 1200km	4346.3	1227.9 5574.2	0.977 0.977 0.995

The excess speed is in km/sec relative to the flyby body.

The Earth distance is in Astronomical Units (1 A.U. = 149,597,871 km) except for the launch and the Earth swingby, when it is given in km height above the Earth's surface (at injection for "launch"). The Sun distances are all in A.U. For flybys, the phase angle is the approach S/C-object-Sun angle, where object is either the asteroid or the Earth. For Delta-V's, it is actually the "fanbeam Sun angle", which is the angle required to orient the spacecraft during the maneuver with the LVA engine so that the Earth is in the fanbeam antenna plane. The value of this angle is calculated for DSM 1 and 2 and Rendezvous 1, and estimated for Rendezvous 2, 3, and 4. The Solar elongation is the Earth-S/C-Sun angle.

- The Pass distance is given in Earth equatorial radii for the launch and Earth swingby, and is given in kilometers for the
- s Delta-V is in meters/second. The Transfer Trajectory Insertion (TTI) delta-V is performed by the Delta 2nd and 3rd stages.

  This is the cosine of the fanbeam Sun angle.
- n c:\near\lwmfinal.tab

### **Comparisons**

Propagation from DSM (July 4, 1997) to Earth swingby perigee (January 22, 1998), comparison of perigee states:

	Δt, sec	Δx, km	Δy, km	Δz, km
Swingby - JPL	0.45	4.6	4.8	4.4
NEAR_Sim - JPL	35	35	12	4

Propagation from transfer trajectory insertion (February 16, 1996) to Mathilde flyby (June 27, 1997), comparison of heliocentric state at the Mathilde flyby:

Swingby - NEAR Sim Max. coordinate difference 1100 km

JPL - NEAR\_Sim Max. coordinate difference 4200 km

Pomer exponent = 2.00, Ref. dist. = 2.1890 A.U., Total delta-V reference for penalty calc. = 1170.0 m/sec d:\hintop2\nearmar1.s6c DMD 95 May 9

\*\*\*\* Case 3 \*\*\*\*
March 1, 1996 2.08298880E+01 G.M.T.
2450144,3679 Julian Date
JDATE,IDATE,DDAYA(LINE),LDAYS,GMT,IEXGMT= March 1, 1996 1996 Mar 1 1.867912000 1 2.0830 1

November 26, 1996 1.701690000E+01 G.M.T.
2450414,2090 Julian Date
JDATE,IDATE,DDAYA(LINE),LDAYS,GMT,IEXGMT= Movember 26, 1996 1996 Mov 26 26.709037500 26 1.7017 1

TARGET HELIOCENTRIC J2000 ECLIPTIC POSITION, IGN= 215541270.3 -228907720.3 231100.0

NELIOCENTRIC J2000 EQUATORIAL POSITION, IGN= 215541270.3 -228907720.3 231100.0

NELIOCENTRIC J2000 EQUATORIAL POSITION, IGN= 215541270.3 -209342721.0 -90509292.4

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JDATE,IDATE,DDAYA(LINE),LDAYS,GMT,IEXGMT= JANUARY 24, 1998 1998 Jan 24 24.185228266 24 4.4655 0

January 24, 1998 4.46547886F400 G.M.T.
2451188.7764 JULIAN Date
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January 10, 1999 6.633860000E+00 G.M.T.
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Movember 26, 1996 1,701690000E+01 G.M.T.
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March 1, 1996 2000 EQUATORIAL POSITION, IGN= 213463459,3 -210913107.3 -91377648.7

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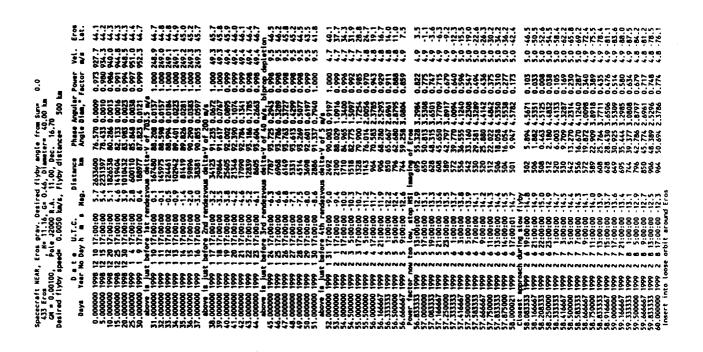
TARGET -E

January 23, 1998 2.26282947E+01 G.M.T. 2450837.4428 Julian Date

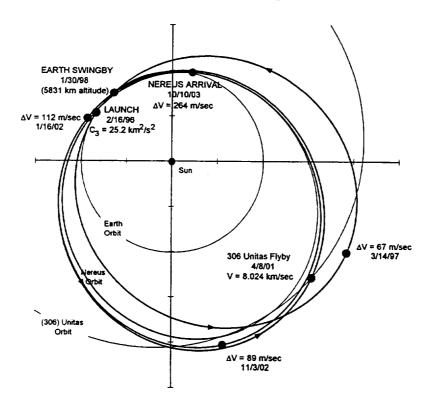
Pass Earth at 7.077 km/sec

Mindist
CLOSE APPROACHES, 1997 3 TO 1998 1, WITHIN 0.10 0.10 A.U. OF NEAR DSM2E

Year	Mo	Day	/ No.		Name	Distance	DelZ	Inc.	DTZ	Diam.	Class	٧	Zdot	Phase	RSun	Delta	Days
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1997		17	4069		Blakee	0.082	0.081	2.21	523.4	10.00		5.1	-0.1	119.0	2.176	3.132	524.5
1997	3	22	177	*	i rma	0.086	-0.009	1.4	28.5	75.30	C:	6.0		68.0			529.5
1997	4	12	2712		1937 YD	0.090	0.004	0.8	28.3	8.71				121.9			551.0
1997	4	13	13540		2140 P-L	0.082	-0.069	10.3	30.0			7.3			2.164		552.0
1997	4	22	12701		1986 VY	0.092	0.017	4.0	-20.3	7.94		6.0			2.059		561.0
1997	4	24	3236		Strand	0.036	0.032		192.3	11.48			-0.3		2.128		562.5
1997	5	8	11		Parthenope	0.215			152.2					138.8			576.5
1997	6	13	11169		1990 TB1	0.060	-0.011		6.3	12.59	•			145.4			612.5
1997	6	14	10764		1981 EM30	0.084	-0.062		141.2					123.0			613.5
1997	6	14	3295		Murakami	0.070			-27.4	18.20				113.3			613.5
1997	6	19	12592		1990 QT2	0.038		0.8	77.9	7.94				118.6			618.5
1997	6	24	253	*	Mathilde		-0.009	6.7	-5.8	61.00				136.4			623.5
1997	7	-4	12504		1990 SA2		0.005		-16.3	7.94		7.7		136.8			633.5
1997	7	17	1422		Stromgrenia	0.084	0.017		26.8	13.06	9	8.0		133.1			647.0
1997	7	22	144		Vibilia		-0.160		408.6	146.00		9.5		146.3			651.5
1997	7	23	10443		1979 SJ	0.056	0.036		-28.0	6.31	•	10.3		144.3			
1997	8		11914		4081 P-L	0.090	0.032	7.1	19.7	7.94				147.8			652.5
1997	_	-	11249		1980 VA		-0.057		-90.2	6.31							663.0
1998		22	,		Earth	0.000	0.000	0.0						135.7			695.5
.,,,	•				cai (II)	0.000	0.000	0.0	-2.0	12/30.32		6.9	U.U	138.6	0.984	0.000	836.0

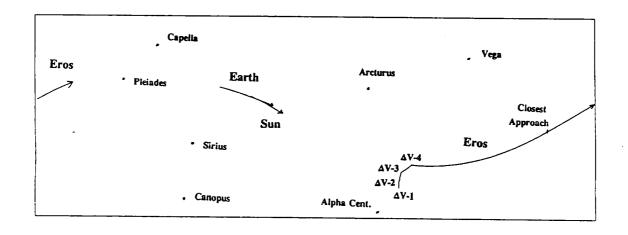


### **NEAR Abort Trajectory Profile**



### The Sky as seen from NEAR during Eros Approch and Slow Flyby

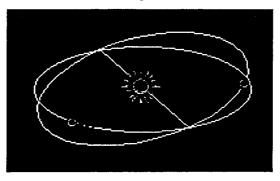
1998 December 10 - 1999 February 8



-		

# **MAnE**<sup>TM</sup>

The Mission Analysis Environment



An Experiment in Modernizing Mission Analysis Software

Jerry L. Horsewood, AdaSoft, Inc.

# **Development Objectives**

- ◆ Provide aerospace community with modern, robust heliocentric mission analysis and optimization tools
  - Promote interest in space exploration
  - Extend mission analysis capability to broader population
  - Enhance opportunities for self study, education
- ◆ Take advantage of recent hardware/software advances
  - Availability, capability and affordability of PC's
  - Graphical user interfaces
  - Multi-processing operating systems
  - Software engineering
- Prevent loss of technology during slack periods of the space exploration program

# Why an Environment?

- Provides ready access to broad range of tools
- ◆ Integration allows convenient sharing of data among components
- Graphical design permits more intuitive and simpler operation
- ♦ Visualization tools enhance comprehension of results
- **◆** Improves productivity
- ◆ Easy to learn

# **Major Components**

- **◆** Trajectory optimization
- ◆ Planetary ephemerides
- ◆ Comet/asteroid ephemerides
- ♦ User defined bodies database
- **◆** Evaluate body locations
- Define dates of body alignments
- ♦ Determine dates of closest approach
- ◆ Evaluate dates of nodal crossings

# Major Components (Cont.)

- ◆ Time/date conversions
- ◆ Solve Lambert's problem
- **◆** Trajectory mapper
- ◆ Personal porkchop plotter
- ◆ Ecliptic projection display
- Mission builder
- ◆ Textual reports
- ◆ 200+ page User Guide

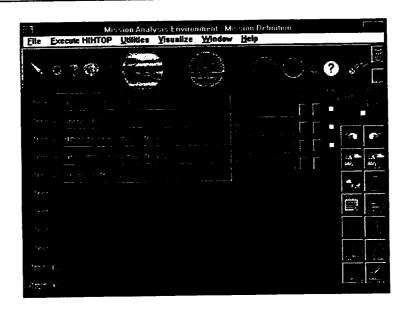
# **Trajectory Optimization**

- ◆ Two-body formulation with optional estimation of velocity losses
- Up to ten heliocentric legs
- Node may be any solar system body or a space burn
- ◆ Final node may be perihelion of arbitrary heliocentric orbit
- ◆ Possible encounters at solar system bodies include:
  - Orbiter or rendezvous with specified stay time
  - Powered or unpowered swingby
  - Powered or unpowered flyby
  - Arrival with or without propulsion maneuver

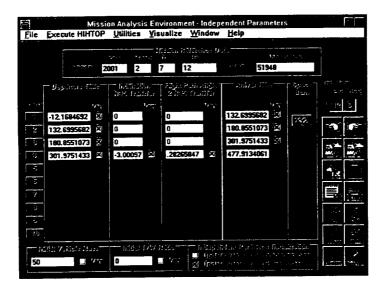
# **Trajectory Optimization (Cont.)**

- Extensive mass and propulsion system models
- Optimization criteria options are:
  - Minimum initial mass
  - Maximum final payload (net) mass
  - Minimum sum  $\Delta v$
  - Minimum mission duration
- **◆** Trajectory leg options supported include:
  - Posigrade or retrograde motion
  - Multi-revolution transfers
  - Transfers that are exact multiples of 180 degrees

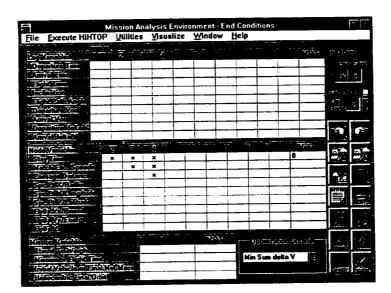
# **Mission Definition Window**



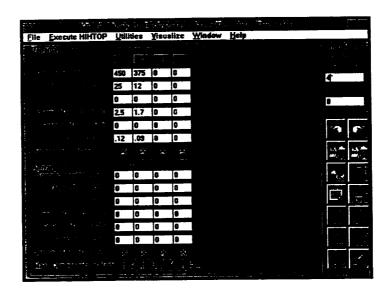
# **Independent Parameters Window**



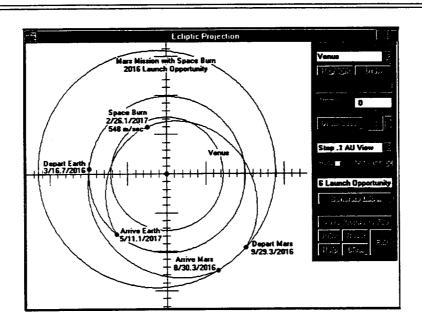
# **End Conditions Window**



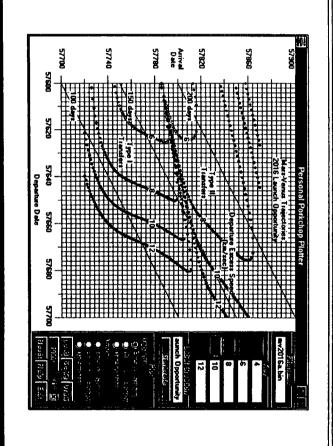
# Mass and Propulsion System Parameters Window



# **Ecliptic Projection View**



# Contour Plotting



•	
	· ·

# - ATMOSPHERIC FLIGHT MECHANICS: Trajectory Optimization Workshop

# GRAVITY ASSIST TARGETING USING

#### SWINGBY A MISSION PLANNING TOOL

By

David C. Folta

**Goddard Space Flight Center** 

Flight Dynamics Division

August 7, 1995



# **SWINGBY**



- INTRODUCTION
- MATHEMATICAL PRINCIPLES
- HARDWARE/SOFTWARE
- SUMMARY



# **INTRODUCTION**



#### **SWINGBY FEATURES**

• Pull Down / Pop-up Menus

• Configurable Menus and Graphical Output

- Systems: File paths and system cmds

- Universe: All planets and user input bodies

- Mission: Configure files, edit, target, run mission

- Configuration: Propagator Config., force and launch models,

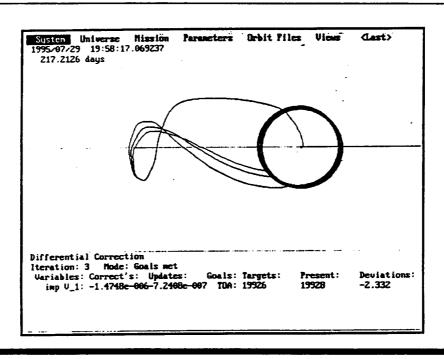
Attitude configuration

- Trajectory: Trajectory and Ephemeris input/output

- Final State: Last Computed State

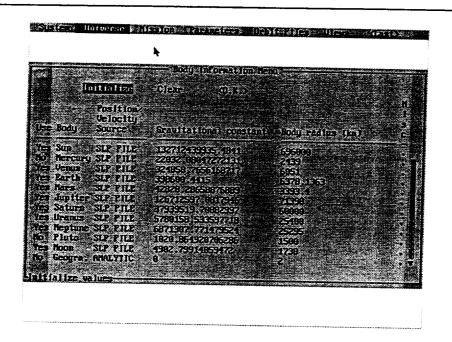














# INTRODUCTION

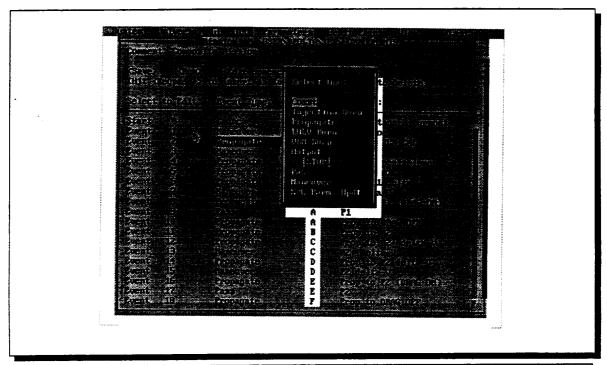


#### **SWINGBY FEATURES**

- Element and Celestial Coordinates Conversion Keplerian/Cartesian/Spherical
- Numerous Coordinate Systems and Object views
- Analytic and File Based Body Ephemerides
- Time Conversions for UTC, UT1, TDT, TDB, and A.1
- Event Driven: Selection of Sequence of Events For Example: Launch, Coast, Maneuver, Propagate, Stop
- Multiple Stopping conditions (logical OR operator) for Propagation Event

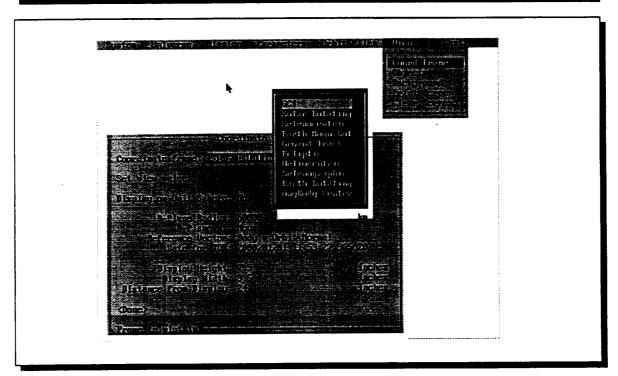






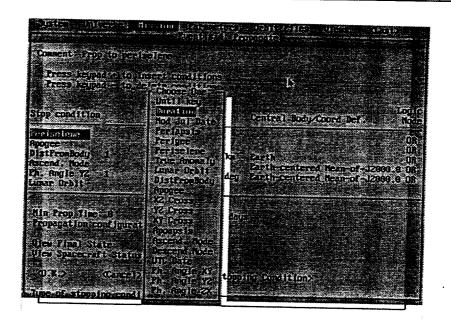
















#### **Trajectory Propagators:**

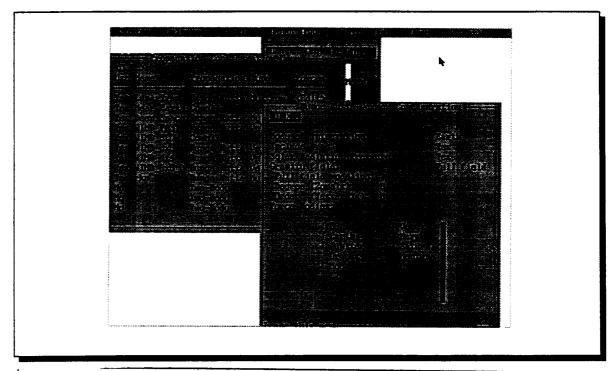
- •Two-Body; Multiconic
- •Runge-Kutta Nystrom (8/6) & Verner(9/8)
- •12th order Cowell Predictor/Corrector
- Adaptive Step Size Control

#### Force Models:

- Solar Radiation Pressure
- Geo and Lunar Potential Models (50x50:21x21)
- Jacchia-Roberts Atmospheric Drag
- Solar/Lunar/Planetary via Universe Selection

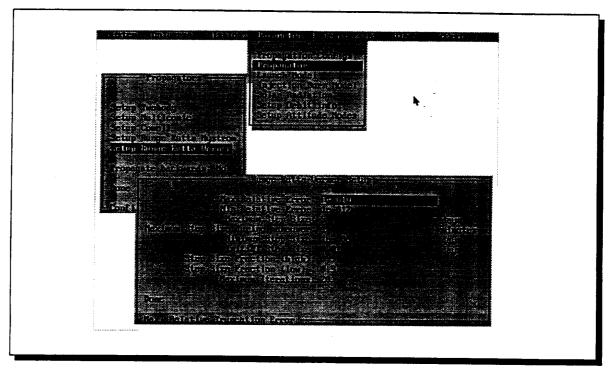
















#### **Targeting Method:** Differential Corrector

- Truncated taylor series solved numerically
- Sensitivity matrix for >1 variable and goal
- Perturbations ,  $\Delta x$  and  $\Delta y$  are user defined

$$f(x + \Delta x, y + \Delta y) = A,$$
  
$$g(x + \Delta x, y + \Delta y) = B.$$

$$\begin{bmatrix} x_1 \\ y_1 \end{bmatrix} \approx \begin{bmatrix} x_0 \\ y_0 \end{bmatrix} + \begin{bmatrix} \frac{\partial f}{\partial x} & \frac{\partial f}{\partial y} \\ \frac{\partial g}{\partial x} & \frac{\partial g}{\partial y} \end{bmatrix}^{-1} \begin{bmatrix} A - f(x, y) \\ B - g(x, y) \end{bmatrix}$$





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# **Optimization Methods: Objective Function**

$$F(u) = \left[ \sum_{i=1}^{n} (W_i [(Z_i(u) - X_i)/S_i])^{P_i} \right]^{P}$$

Where W is weight, S is scale factor, and P is power



# **Mathematical Principles**



#### Optimization Methods: Steepest Descent

- Compute gradient and Hessian using central differencing for search direction
- Line search for minimum assuming quadratic and cubic fits
- Objective function calculated at minimum values
- Yields 1 dimensional minimizer, used as initial point for next iteration
- Procedure repeated until gradient falls below given value





## Optimization Methods: Quasi-Newton

- •Compute gradient and Hessian Matrix using BFGS algorithm for search direction
- •Line search for minimum assuming quadratic fit
- •Objective function calculated at minimum values
- •Result used as initial point for next iteration
- •Procedure repeated until gradient falls below given value





```
Angle of rotation & Section of the s
```





#### Maneuver Modeling:

Impulsive and Finite  $\Delta Vs$  based on fixed and user defined body attitude systems

- Attitude Examples:
   LVLH: VBN: SUN Pointing: Inertial: User Defined
   Body / Reference / Attitude / Inertial Coordinates
- Attitude input as Euler angles in user defined coordinates

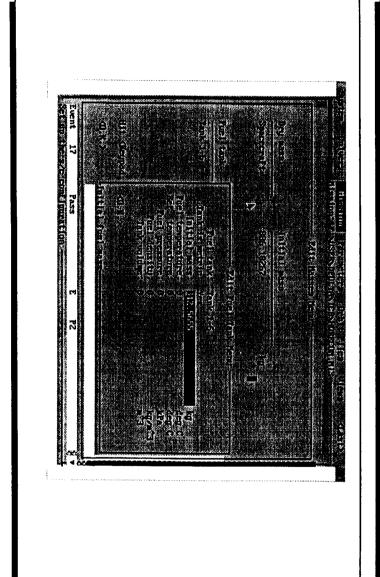


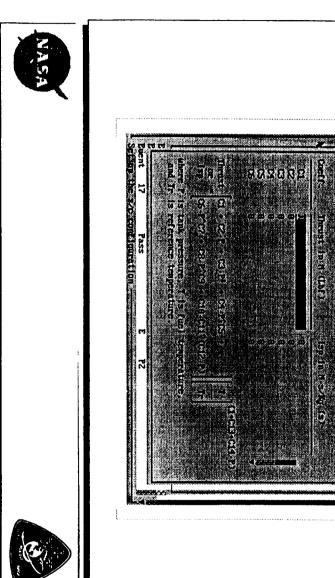
# **Mathematical Principles**



#### **Maneuver Modeling**

- Analytic Equation for engine model (Thrust and ISP) based on performance coefficients, pressure, temperature
- Fuel Tank Parameters for Mass Property Calculations S/C dry and wet mass, fuel density, volume
- Targeting on Duration or ΔVusing initial impulsive ΔV or first guess



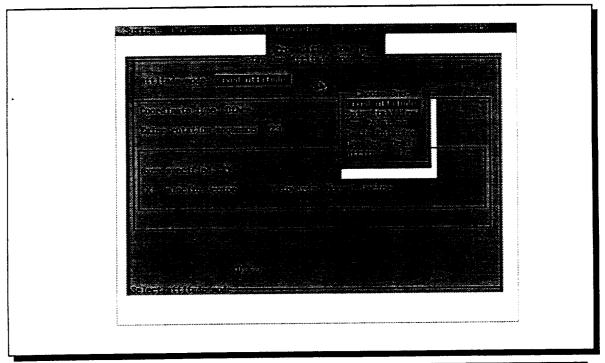






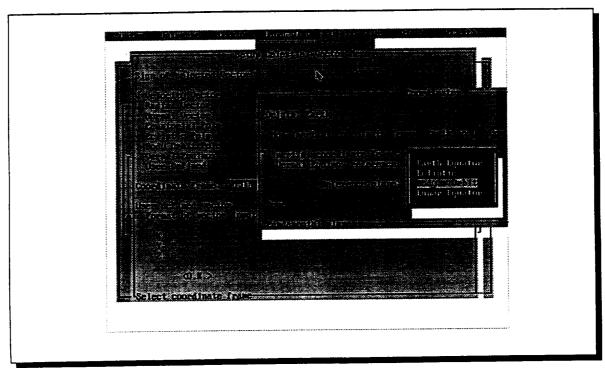














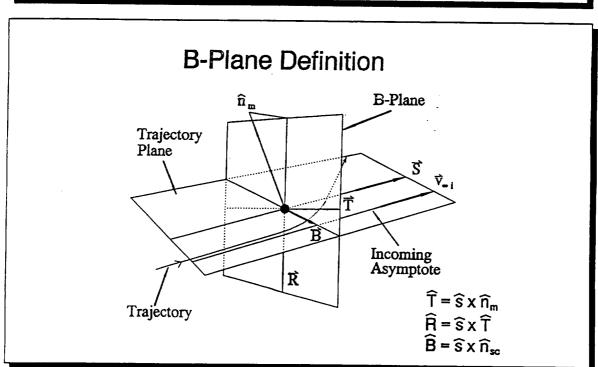


#### **B-Plane and Floating Endpoint Targeting**

- Analytical Method to determine B-Plane targets to achieve desire gravitational assist.
- B-Plane will change slightly with each spacecraft trajectory update
- Floating Endpoint coverts B-Plane targets to Keplerian Targets, e.g. B.T and B.R become persilene distance and lunar inclination

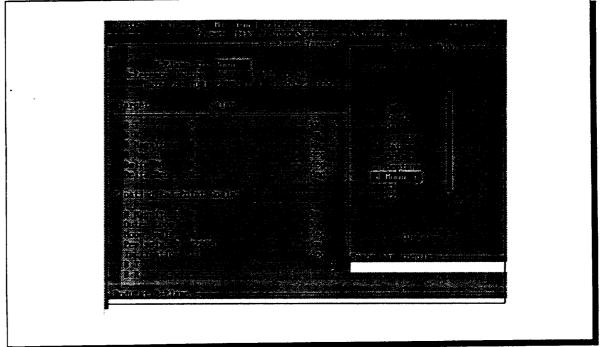






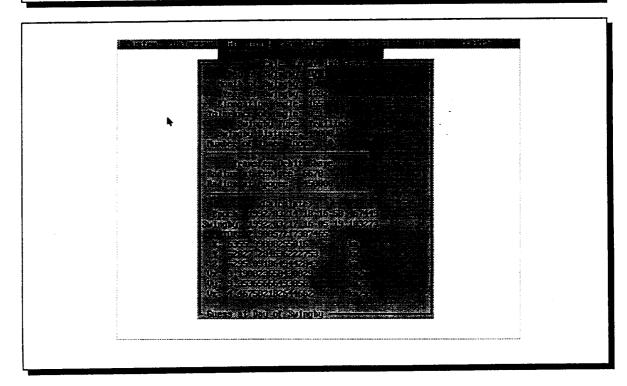














# **Description**



#### **TARGETING OF MISSIONS**

Clementine: Vary Launch; Coast; TTI DV, and VBN DVs to

Achieve B-Plane and Floating End-Point Lunar Goals; Orbital elements for Mapping; and B-Plane

for Asteriod Targeting

WIND: Vary Orbital Elements and VBN DVs to Achieve

Desired Angular Separation of Moon and S/C; B-Plane Parameters; and RLP at Plane Crossings

SOHO/ACE: Vary Launch/Coast, Elements, VBN DVs, to Achieve

**RLP Targets** 



# Software and Hardware



#### Hardware Environment:

386/486 with  $\sim 4$  meg memory

EGA/VGA

~1 meg disk space

Designed for LAN

#### **Source Code:**

C and Fortran

~125000 LOC



## **SUMMARY**



- SWINGBY is an Interactive tool with graphical capabilities which provides unambiguous interpretation of Complex Trajectories.
- SWINGBY has been effectively used for all gravity assist, libration point, comet, and asteroid mission analysis and support.
- SWINGBY high fidelity models have been proven through both operations and comparisons to other numerical methods and Software, (ASAP, GMAS, GTDS).
- SWINGBY is being used for educational purposes at several Universities and by the Commercial world.



Soals: offn: offic: TOR:

WIND: Phasing loops; Lunar swingby to the interior Lagrange point

With Swingby, the user can create paper copies, store data in electronic files, and change the

Swingby was used to plan and support the Clementine mission. It is being used to plan and support the WIND, SOHO and ACE missions as well. Swingby has been designed for prelaunch

inalysis and design, and for maneuver planning and redesign during a mission

missions that use lunar gravity assists, halo orbits, comet intercepts, and missions to other planets

The user interacts with Swingby using pulldown menus and displays of spacecraft trajectories on

the screen. These trajectories can be displayed in a number of

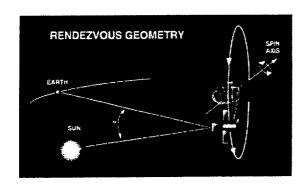
geocentric inertial coordinates, Moon-centered

This feature allows the user to understand how the trajectories are affected when

changes are made in the properties of the spacecraft and its environment. The rapid feedback also Swingby displays graphical and numerical information as it is calculated to give the user quic

# NEAR-Sim Near Earth Asteroid Rendezvous Simulation

{Presented at the AIAA Workshop on Trajectory Optimization Methods and Applications}



#### **NEAR-Sim** Development Team

Victor R. Bond / MDA Gregory A. Clubb / MDA Michael F. Fraietta / MDA Robert G. Gottlieb / MDA Steven J. Sponaugle / MDA

David W. Dunham / APL James V. McAdams / APL

> clubb@pat.mdc.com Voice: (713) 244-4433

> > 7 August 1995

MDA - Houston Division !

#### Contents

- □ Background
  - The Near Earth Asteroid Rendezvous Mission
  - The Advanced Simulation Development System (ASDS)
  - The Near Earth Asteroid Rendezvous Simulation (NEAR-Sim)
- ☐ Approach for **NEAR-Sim** Trajectory Convergence
  - Overlapped-Conic Starter<sup>1</sup> with BG14 Calibration<sup>2</sup>
  - Precision Split Trajectory Convergence using State Transition Matrix
- □ **NEAR-Sim** Results
  - Overlapped-Conic Targeting Results
  - State Transition Matrix Targeting Results
- □ Summary
- 1. Overlapped-conic targeting based on work by Sam Wilson/JSC-retired [Ref. TRW 5521.4-18, May 1971]
- 2. Bond & Gottlieb 14-Element Precision Propagation Method (Ref. JSC-23495, May 1989)

■ NEAR\_Sim Development Team

7 August 1995 ===

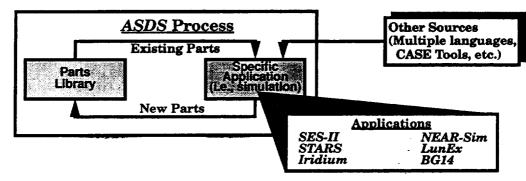
# Background

- ☐ The Near Earth Asteroid Rendezvous (NEAR) mission is scheduled to be launched in February 1996 by the Applied Physics Laboratory (APL) of Johns Hopkins University (Laurel, Maryland)
- ☐ This mission will orbit an 805 kg satellite for approximately one year around the 13 x 15 x 36 km asteroid Eros, starting in February 1999
- ☐ This mission is one of the first two missions in NASA's Discovery class series, and will be launched by a McDonnell Douglas Delta rocket in February 1996
- ☐ In 1994, McDonnell Douglas Houston Division was tasked to provide APL with an *ASDS* developed precision trajectory targeting application which we call *NEAR-Sim*
- ☐ NEAR trajectories presented here were used only for the testing of the software
  - The actual planned NEAR trajectory will be given in the Oct-Dec '95 Journal of the Astronautical Sciences by R. Farquhar, D. Dunham, and J. McAdams, APL, "Mission Overview and Trajectory Design"

NEAR\_Sim Development Team

7 August 1995

#### What is **ASDS**?



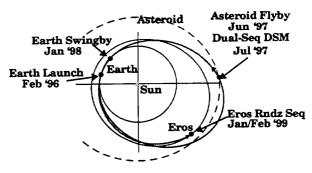
- $\Box$  The Advanced Simulation Development System (ASDS)
  - A generic, powerful, yet simple application framework
  - Unique feature facilitates rapid development of any simulation
  - Independent of programming language
- ☐ A library of **reusable** and **integrable** building blocks
  - From high-level executive parts to low-level primitive parts
- ☐ A repeatable process which leverages reuse of requirements, design, code, testing, and documentation

NEAR\_Sim Development Team :

7 August 1995 =

#### What Is **NEAR-Sim**?

- □ Near Earth Asteroid Rendezvous Simulation (NEAR-Sim)
  - 1994 Applied Physics Lab (APL) Task Version-1 built in 4 months
    - Converges NEAR trajectory using Deep Space Maneuver and an Earth gravity assist to setup the final Eros rendezvous sequenced maneuvers
  - 1995 APL Task Version 2 delivered in June 1995
    - · Dual sequenced deep space maneuver
    - Pre and Post Earth swingby mid-course correction capability
    - Deep Space Network (DSN) AOS/LOS/Maximum elevation events
- ☐ NEAR-Sim was built in ASDS using existing parts:
  - Precise (over-lapped conic targeting)
  - **BG14** (precise calibration)
  - Fast-Phi (state transition matrix targeting)

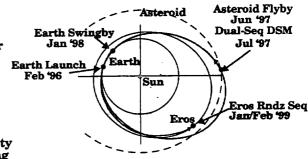


NEAR Sim Development Team :

7 August 1995

# **NEAR-Sim** History

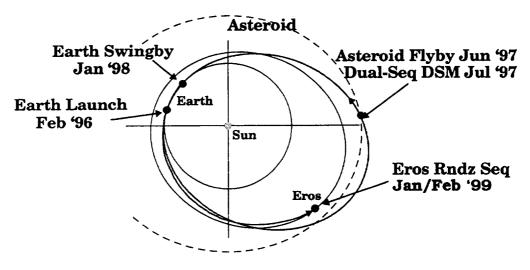
- □ NEAR-Sim Features include:
  - JPL Constants and Dastcom\_3 Asteroid/Comet Ephemeris Packages in ASDS Version 3.4
  - NEAR mission precise convergence using impulsive ΔV or finite thrust for the two major post-injection maneuvers
    - Deep Space Maneuver (DSM)
    - Eros Rendezvous Maneuver (up to 10 sequenced maneuvers)
  - NEAR parametric study capability using over-lapped conic targeting



- □ NEAR-Sim Version-0 was electronically delivered to APL
  - Version-0 delivery on 20 July 1994 for the purpose of customer feedback
- □ NEAR-Sim Version-1 was delivered to APL in September 1994
- ☐ NEAR-Sim Version-2 was delivered to APL in June 1995

# **NEAR-Sim** Approach & Results

- □ NEAR-Sim requires Earth injection state vector and time
- $\square$  **NEAR-Sim** provides precise DSM and Eros sequenced maneuver rendezvous computations with impulsive  $\Delta V$  or finite thrust



■ NEAR\_Sim Development Team

7 August 1995 🚍

# **NEAR-Sim** Technical Approach

- ☐ Two Pronged Approach
  - Use overlapped conic targeting coupled with **BG14** calibration to get very close to the precision trajectory
  - Use State Transition Matrix (STM) targeting coupled with the BG14 equations of motion for NEAR trajectory convergence
- ☐ The NEAR trajectory is extremely sensitive to perturbations
  - Must model high order Earth gravity and all planets except Pluto, as well as solar radiation pressure
- Experience indicated that the sensitivity could be overcome by splitting the mission into two parts and converging the trajectory at Earth swingby
  - Overlapped conic solutions for both parts with an iteration to force the incoming/outgoing perigee velocity magnitudes to be very nearly equal
  - Precision trajectories are integrated forward from the deep space maneuver (DSM) and backwards in time from Eros (both to perigee)

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# Overlapped Conic Targeting

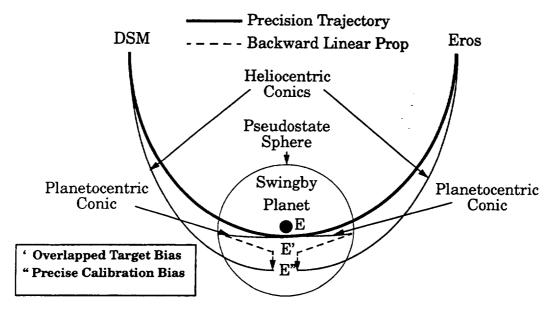
- ☐ Plane of the swingby orbit is defined by the V vectors of the two conics from the DSM to Earth and from Eros to Earth
- ☐ Two conic solutions are obtained that include a bias provided by integration of the equations of motion of the system using **BG14**
- These two solutions will not (in general) have the same perigee velocity or the same perigee phase angle
- The velocities are converged by varying the time of perigee passage
- ☐ The perigee phase angles are converged by varying the altitude of perigee passage
- ☐ The overlapped-conic Earth swingby time and altitude are used to initialize the precision computations

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# Schematic of Targeting Technique

☐ Overlapped Conic with BG14 Precision Calibration



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# **Uncalibrated Overlap Results**

DSM -to- Eros:

3 July 97 -to- 6 Feb 99

Perigee Swingby:

Mid-Point @ 500 km swingby altitude (first guess)

Table 1: Overlap Swingby Convergence Results

Total Iter	Alt Iter	TP Iter	Swingby Date 1998	Position RSS Error (km)	Velocity Magn Error (m/s)	Perigee Phase Error
Nom	Nom	Nom	1/1 00:00:00.00	1782.947	2562.929	-0.257031
#1	0	1	1/1 01:00:00.00	1782.573	2558.006	-0.256978
#2	0	2	1/22 16:36:05.43	23.845	26.788	-0.003434
#3	0	3	1/22 22:06:02.08	17.945	-7.293	0.002568
#4	0	4	1/22 20:55:25.79	9.340	0.038	0.001276
#5	1	4	1/22 20:55:25.79	315.240	-0.059	0.042714
#6	2	4	1/22 20:55:25.79	3.215	0.041	-0.000053
#7	3	4	1/22 20:55:25.79	3.195	0.041	0.000002

Converged Swingby: 22 Jan 98 20:55:25.79 @ 485.2407 km swingby altitude

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# State Transition Matrix (STM) Targeting

- ☐ Given an initial date and an arrival date, the initial position (DSM) and the final position (Eros) are known
- $\supset$  The problem is to find the initial velocity  $(V_i)$  and the final (arrival) velocity  $(V_a)$  that cause the state at the terminus of the two trajectories (perigee) to be equal
  - Perigee states from the forward and backward legs must be equal

$$X\!\!\left(t_i,t_p,P_i,V_i\right)_{\!\!f}=X\!\!\left(t_p,t_a,P_a,V_a\right)_{\!\!b}$$

 $\Box$  Given a nominal trajectory and the state transition matrices relating  $t_i$  and  $t_p$ , and  $t_a$  and  $t_p$ , we require

$$X_f + \delta X_f = X_b + \delta X_b \Rightarrow X_f + \frac{\partial X_f}{\partial V_i} \delta V_i = X_b + \frac{\partial X_b}{\partial V_a} \delta V_a$$

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# STM Targeting (cont'd)

 $\hfill\Box$  The solution for  $\delta V_i$  and  $\delta V_a$  can be found from

$$\begin{bmatrix} \delta V_i \\ \delta V_a \end{bmatrix} = \begin{bmatrix} -\frac{\partial X_f}{\partial V_i} \frac{\partial X_b}{\partial V_a} \end{bmatrix}^{-1} \begin{bmatrix} X_f - X_b \end{bmatrix}$$

- The iterative process continues until the difference between  $X_f$  and  $X_b$  is small,  $|P_f P_b|/|P_f| + |V_f V_b|/|V_f| < 10^{-8}$
- ☐ The advantage of starting with the calibrated overlapped solution is that only a few iterations are required to converge the precision trajectory (generally < 5)
- ☐ The combination of the state transition matrix and **BG14** was first developed in the **Fast-Phi** application in **ASDS** (developed for the Johnson Space Center in 1992)

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#### **BG14** - STM Combination

- □ BG14 is a 14 element variation of parameters algorithm which allows exceptionally rapid and accurate propagation of the state
- $\Box$  The independent variable for **BG14** is fictitious time, s, defined by the Sundman transformation

$$\frac{dt}{ds} = r$$

□ Since the cartesian state generated by **BG14** satisfies Newton's laws, it is still possible to compute a 6x6 state transition matrix even though the "state" of **BG14** contains 14 elements

Ξ

# **BG14** - STM Combination (cont'd)

The standard cartesian state transition matrix can be integrated simultaneously with the **BG14** equations provided they are modified to be

$$\Phi' = \frac{d\Phi}{ds} = \frac{d\Phi}{dt}\frac{dt}{ds} = r\frac{d\Phi}{dt} = r\frac{\partial \dot{X}}{\partial X}\Phi$$

- The integration of φ' is accurate because they are variational equations about a given (albeit sensitive) trajectory, and because the real escalation in sensitivity would occur after Earth swingby (which never occurs)
- ☐ The following STM results were computed by **NEAR-Sim** using the overlapped-conic derived Earth swingby time and altitude as input

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## Calibrated STM-with-Thrust Results

☐ Propulsive State Transition Matrix Targeting

DSM -to- Eros:

3 July 97 -to- 6 Feb 99

Overlap Swingby:

22 Jan 98 20:55:25.79 @ 485.2407 km altitude

Calibrated Swingby:

Prior to propulsive STM targeting, *NEAR-Sim* required 1 calibrated overlapped iteration to clean-up 1.4 m/s velocity error plus 1 calibrated overlapped iteration to clean-up a small mass difference due to stale  $\Delta V$ 's

Table 2: Precision Swingby Convergence Results with Thrust

Iterations	Position RSS Error (m)	Velocity RSS Error (m/s)	Normalized Error
Nom	321393.587	220.589	0.063902805
#1	310.343	0.753	0.000104044
#2	29.688	0.026	0.00006393
#3	5.445	0.004	0.000001140
#4	0.180	0.000	0.00000037
#5	0.031	0.000	0.000000006

■ Precise Swingby:

22 Jan 98 21:08:37.78 @ 485.2407 km altitude

Mathilde Flyby

27 June 97 07:57:57.47 - Close approach of 43.3 km, velocity 9.94 m/s

■ Sparc-4 CPU Time:

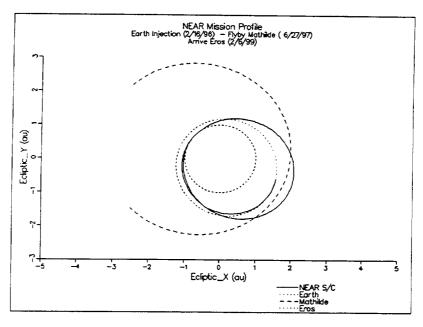
~110 sec (includes final end-to-end precision target-miss run)

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■ Target Miss Results: ~10.5 m and 0.0005 m/s RSS position & velocity errors

# Calibrated STM-with-Thrust Plots

Generated from **NEAR-Sim** plot file created during final end-to-end integration (Earth Injection -to- Eros Rendezvous)

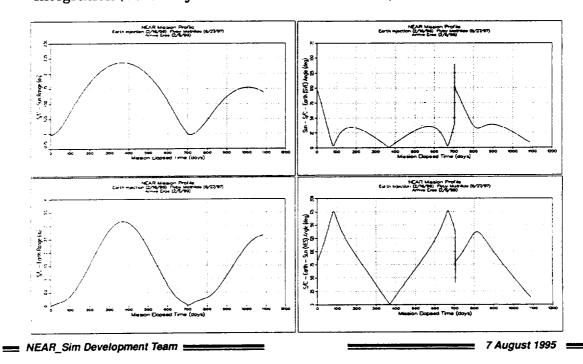


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## Calibrated STM-with-Thrust Plots

☐ Generated from **NEAR-Sim** plot file created during final end-to-end integration (Earth Injection -to- Eros Rendezvous)



## Summary

Determined that this problem is <u>highly</u> sensitive to perturbations

 Must model high order Earth gravity and all planets except Pluto, as well as solar radiation pressure
 Successfully simulated the NEAR trajectory as a split trajectory joined at Earth swingby using *NEAR-Sim* Successfully converged the NEAR trajectory using a combination of overlapped-conic and state transition matrix targeting techniques coupled with the *BG14* precision propagation method
 Demonstrated the applicability of *ASDS* to this problem and Discovery class missions in general

# Development of a MATLAB Toolbox for Trajectory Optimization and Planning

Mark L. Psiaki, Mechanical and Aerospace Engineering Cornell University, Ithaca, N.Y.

August 7, 1995

# Acknowledgments

The Lady Davis Fellowship Trust; The Technion, Haifa, Israel; Moishe Guelman and Josef Shinar.

## **Outline**

- 1. Objectives and problem definition.
- 2. Design and functionality of tool.
- 3. User interface.
- 4. Example problems and solutions.
- 5. Dynamic feasibility problem.
- 6. Cost-constrained dynamic feasibility problem.
- 7. Computational experience.
- 8. Conclusions and future plans.

# **Objectives of Effort**

- Easy-to-learn/use MATLAB tool for numerical nonlinear trajectory optimization.
- Inclusion of research results in constrained LQR problems.
- Easily modifiable testbed for alternative nonlinear programming core algorithm features.
- Tool for dynamic feasibility problems.

# **Mayer-Type Problem Form**

find: u(t) and x(t) for  $t_0 \le t \le t_f$  to minimize:  $J = V_f\{x(t_f), u(t_f), t_f\}$  subject to:  $x(t_0)$  given or  $\phi_{0E}\{x(t_0)\} = 0$   $dx/dt = f\{x(t), u(t), t\}$   $c_E\{x(t), u(t), t\} = 0$   $c_I\{x(t), u(t), t\} \le 0$   $\phi_{fE}\{x(t_f), u(t_f), t_f\} = 0$   $\phi_{fI}\{x(t_f), u(t_f), t_f\} \le 0$   $du_i/dt = 0$  for  $i = (m_v + 1), \dots, m$ 

# **Optimization Techniques**

• Direct Nonlinear Programming Method.

Piecewise-linear u(t)

4th-order Runge-Kutta integration of dynamics

- Discrete-Newton Hessian approximation.
- Modified SQP search direction computation.
- Curved line search with penalty function or augmented Lagrangian function as merit function.
- Cost-constrained procedure to generate a good guess.

# **Features of Toolbox**

- Problem modeling via user-programmed functions and first-derivatives of problem functions.
- Different problem form options.
- Optimization of fixed parameters;  $du_i/dt = 0$ .
- No need of special modeling/analysis for statevariable inequality constraints or singular arcs.
- Partial constraint/partial optimization of  $x(t_0)$ .
- Optional feasibility phase.

# Features of Toolbox (Cont'd.)

- User-selected number of Runge-Kutta steps per time node.
- Auto-selection of finite difference intervals for discrete-Newton Hessian approximation.
- Auto-checking of user-programmed derivatives.
- Auto-scaling of constraints.

# **Typical MATLAB Call of Tool**

[xhist,uhist,J] = trajopt0(x0',uhistges,thist,... 'f\_fn','Vf\_fn','c\_fn','phif\_fn','phi0\_fn',options)

Names in MATLAB Terms in problem statement

xhist, uhist, uhistges x(t), u(t), first guess of u(t)

thist  $[t_0;t_1;t_2;...;t_N]$  where  $t_N \equiv t_f$ 

 $f_fn$ ,  $Vf_fn$ , etc. .m-file names for  $f\{x(t), u(t), t\}$ ,

 $V_{\rm f}\{x(t_{\rm f}),u(t_{\rm f}),t_{\rm f}\}$ , etc.

options Input switches, parameters.

# 2-D Minimum-Time to Origin Problem

find:  $t_f$  and  $\mathbf{a}(t)$  for  $0 \le t \le t_f$ to minimize:  $J = t_f$ subject to:  $\mathbf{x}(0)$  and  $\mathbf{v}(0)$  given  $d\mathbf{x}/dt = \mathbf{v}$  $d\mathbf{v}/dt = \mathbf{a}$  $\mathbf{a}^T(t) \mathbf{a}(t) \le 1$  $\mathbf{x}(t_f) = \mathbf{0}$  $\mathbf{v}(t_f) = \mathbf{0}$ define:  $\mathbf{x} = [\mathbf{x}; \mathbf{v}], \ \mathbf{u} = [\mathbf{a}; t_f], \ \tau = t/t_f, \ f(\mathbf{x}, \mathbf{u}, \tau) = d\mathbf{x}/d\tau$ 

# 2-D Minimum-Time to Origin f\_fn.m File

function 
$$[f, dfdx, dfdu] = f_fn(x, u, \tau, gradflag)$$
  
 $f = [x(3); x(4); u(1); u(2)]*u(3);$   
if  $gradflag = 1$   
 $dfdx = zeros(4,4); dfdu = zeros(4,3);$   
 $dfdx(1,3) = u(3); dfdx(2,4) = u(3);$   
 $dfdu(3,1) = u(3); dfdu(4,2) = u(3);$   
 $dfdu(3,3) = [x(3); x(4); u(1); u(2)];$   
end

# Maximum Range in Unpowered Flight

find:  $t_f$  and  $C_L(t)$  for  $0 \le t \le t_f$ 

to min.:  $J = -x(t_f)$ 

subject to: v(0),  $\gamma(0)$ , h(0), x(0) given

 $dv/dt = -D[\rho(h), v, C_L, M]/m - g \sin\gamma$ 

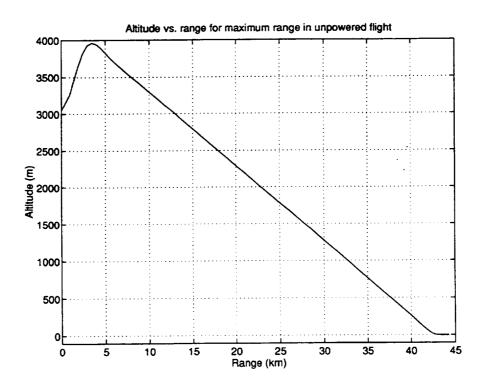
 $d\gamma/dt = \{L[\rho(h), v, C_1]/m - g \cos\gamma\}/v$ 

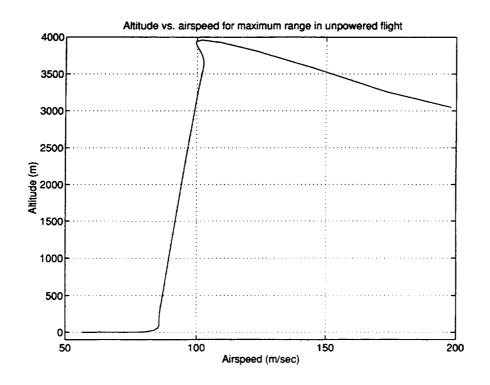
 $dx/dt = v \cos \gamma$ ,  $dh/dt = v \sin \gamma$ 

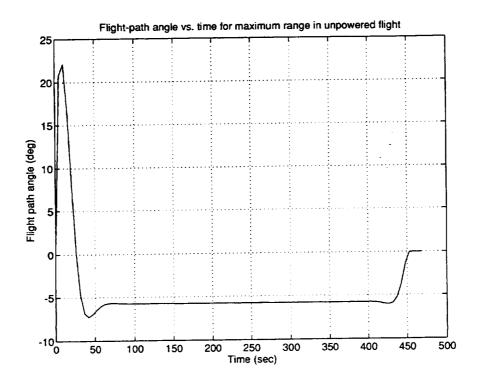
 $\boldsymbol{C_L} \leq \boldsymbol{C_{Lmax}}(\boldsymbol{M}), \ L[\rho(\boldsymbol{h}), \boldsymbol{v}, \boldsymbol{C_L}]/mg \leq \boldsymbol{n_{zmax}}$ 

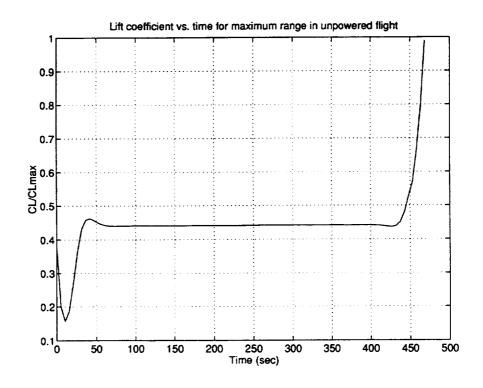
 $h \ge 0$ 

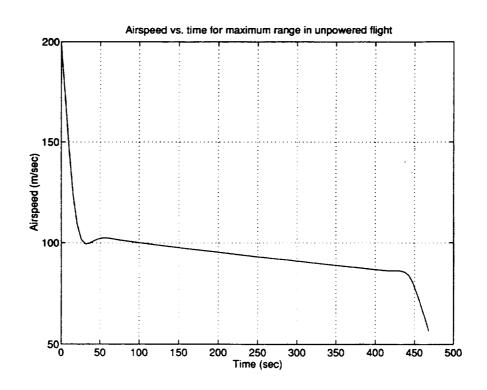
$$v(t_f) \ge v_{stall}, \ \gamma(t_f) = 0, \ h(t_f) = 0$$











# Minimum-Propellant-Mass Electric Propulsion Orbit Transfer with High Drag

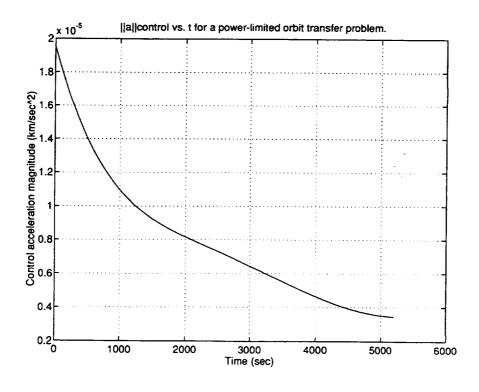
find:  $t_f$  and  $[a_r(t), a_{\theta}(t)]$  for  $0 \le t \le t_f$ 

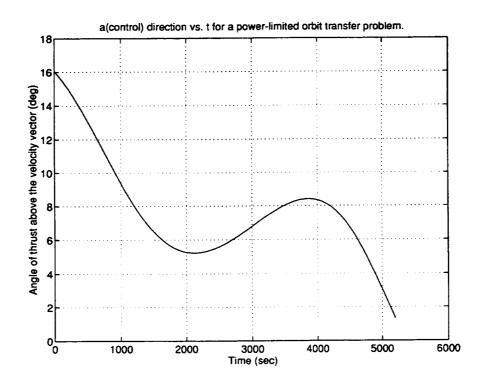
to min.:  $J = \int_0^{tf} (a_r^2 + a_\theta^2) dt$ 

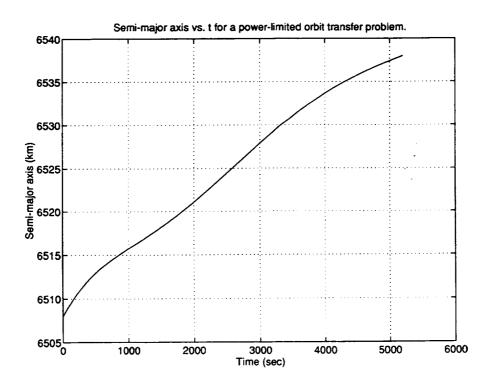
subject to: r(0),  $dr/dt|_0$ ,  $\theta(0)$ ,  $d\theta/dt|_0$  given

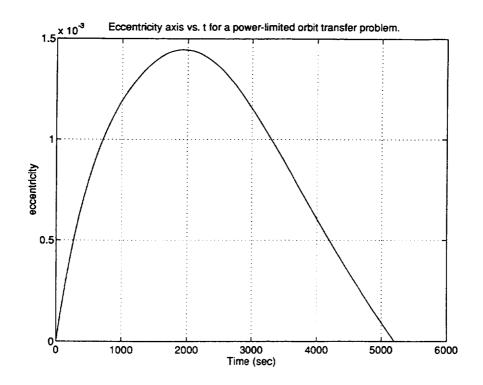
 $d^{2}r/dt^{2} - r(d\theta/dt)^{2} + \mu_{E}/r^{2} = a_{r} - (a_{D})_{r}$   $r d^{2}\theta/dt^{2} + 2(dr/dt)(d\theta/dt) = a_{\theta} - (a_{D})_{\theta}$   $a_{D} = \rho(r)[(dr/dt)^{2} + r^{2}(d\theta/dt)^{2}] SC_{D}/(2m)$ 

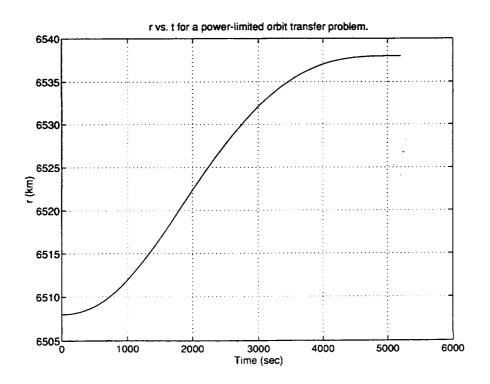
 $r(t_f)$ ,  $dr/dt|_{tf}$ , and  $d\theta/dt|_{tf}$  given;  $t_f \le t_{fmax}$ 

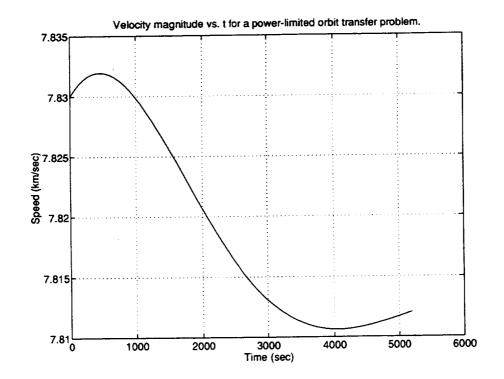












# **Minimum-Time Horizontal Interception**

find:  $t_f$  and  $[T(t), tan \mu(t)]$  for  $0 \le t \le t_f$ 

to min.:  $J = t_f$ 

subject to: v(0),  $\chi(0)$ ,  $\chi(0)$ , y(0) given

 $dv/dt = \{T - D[v,tan\mu,M]\}/m$ 

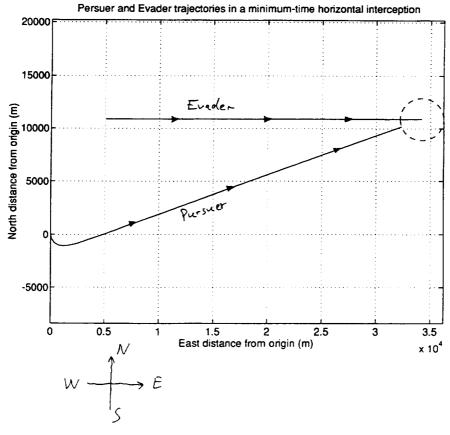
 $d\chi/dt = g \tan \mu/v$ 

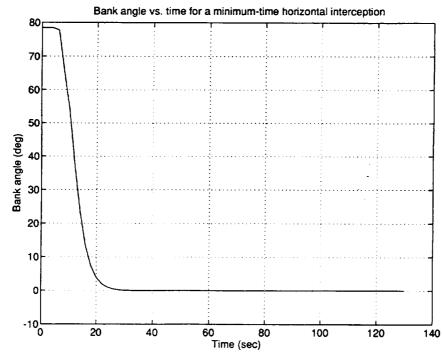
 $dx/dt = v_E - v \cos \chi$ ,  $dy/dt = -v \sin \chi$ 

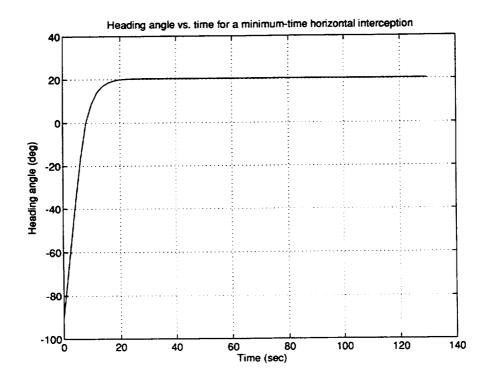
 $\tan^2 \mu \le (n_{zmax})^2 - 1$ 

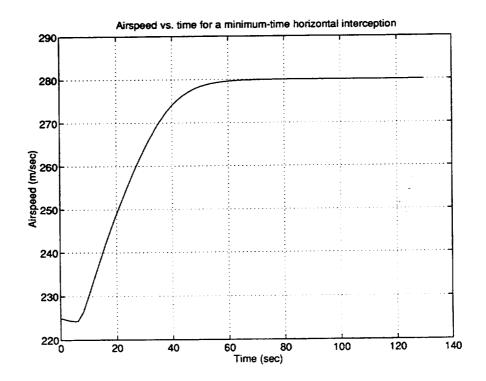
 $\tan^2 \mu \le \{ [\rho v^2 SC_{Lmax}(M)] / [2mg] \}^2 - 1$ 

 $x^{2}(t_{f}) + y^{2}(t_{f}) = r_{f}^{2}$ 









# **Dynamic Feasibility Problem**

find any:

u(t) and x(t) for  $t_0 \le t \le t_f$ 

to satisfy:

 $x(t_0)$  given

 $dx/dt = f\{x(t), u(t), t\}$ 

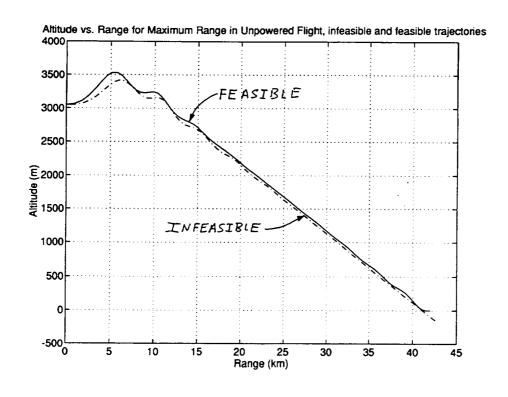
 $c_{\rm E}\{x(t),u(t),t\}=0$ 

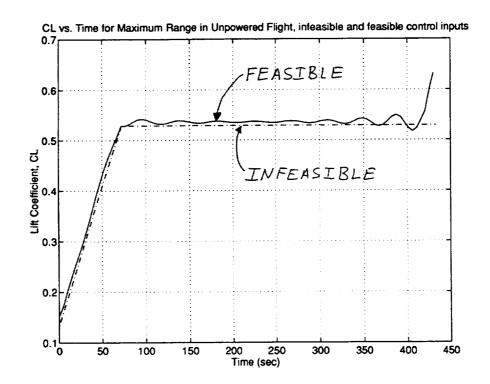
 $c_{\mathsf{I}}\{x(\mathsf{t}),u(\mathsf{t}),\mathsf{t}\}\leq 0$ 

 $\phi_{fE}\{\boldsymbol{x}(t_f),\boldsymbol{u}(t_f),t_f\}=\boldsymbol{0}$ 

 $\phi_{\mathrm{fI}}\{\boldsymbol{x}(t_{\mathrm{f}}),\boldsymbol{u}(t_{\mathrm{f}}),t_{\mathrm{f}}\}\leq\boldsymbol{0}$ 

Note: use same function call but replace 'Vf\_fn' by [].





# **Cost-Constrained Dynamic Feasibility Problem**

find any: u(t) and x(t) for  $t_0 \le t \le t_f$ 

to satisfy:  $V_{f}\{x(t_{f}), u(t_{f}), t_{f}\} \leq J_{target}$ 

 $x(t_0)$  given

 $dx/dt = f\{x(t), u(t), t\}$ 

 $c_{\rm E}\{x(t),u(t),t\}=0$ 

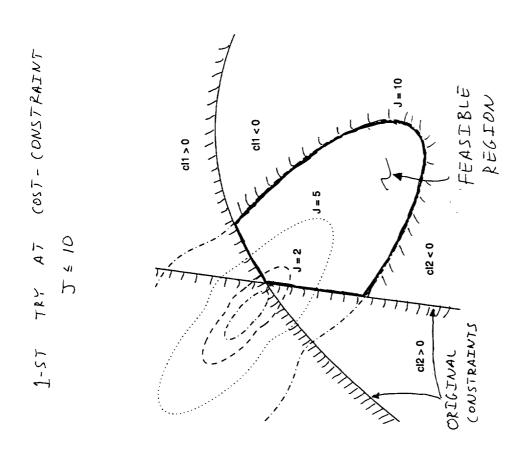
 $c_{\mathsf{I}}\{x(\mathsf{t}),u(\mathsf{t}),\mathsf{t}\}\leq 0$ 

 $\phi_{fE}\{\boldsymbol{x}(t_f),\boldsymbol{u}(t_f),t_f\}=\boldsymbol{0}$ 

 $\varphi_{fI}\{\textbf{\textit{x}}(t_f), \textbf{\textit{u}}(t_f), t_f\} \leq \textbf{0}$ 

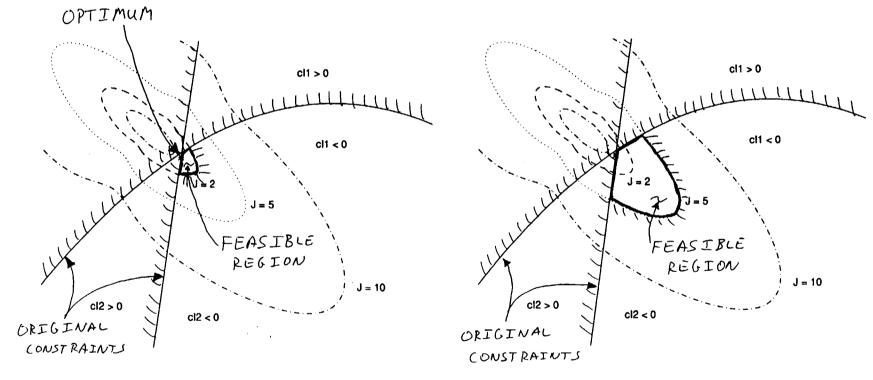
# **Optimization by Cost-Constrained Sequence**

- 1. Set j = 0. Solve non-cost-constrained feasibility problem. Get  $[x^j(t), u^j(t)]$  and  $J^j = V_f\{x^j(t_f), u^j(t_f), t_f\}$ .
- 2. Set j = j+1. Pick  $J_{target}^{j} < J_{target}^{j-1}$ . Solve cost-constrained feasibility problem using  $[x^{j-1}(t), u^{j-1}(t)]$  as first guess.
- 3. If  $[x^{j}(t), u^{j}(t)]$  is feasible and Step 2 converged rapidly, set  $J^{j} = V_{f}\{x^{j}(t_{f}), u^{j}(t_{f}), t_{f}\}$  and go to Step 2.
- 4. If  $J^{j-2} J^{j-1}$  is "small" terminate.  $[x^{j-1}(t), u^{j-1}(t)]$  is near optimum. Otherwise, set j = j-1 and return to step 2, but pick a larger  $J^{j}_{target}$  this time.



3-RD TRY AT COST CONSTRAINT 2-ND TRY AT COST-CONSTRAINT J = 2

J 55



# Maximum Rocket Altitude (Goddard)

find:  $t_f$  and T(t) for  $0 \le t \le t_f$ 

to min.:  $J = -h(t_f)$ 

subject to: h(0), v(0), m(0) given

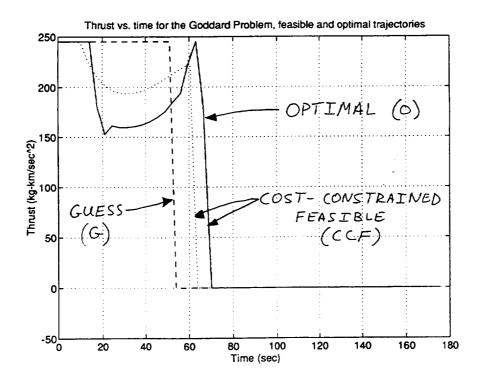
dh/dt = v

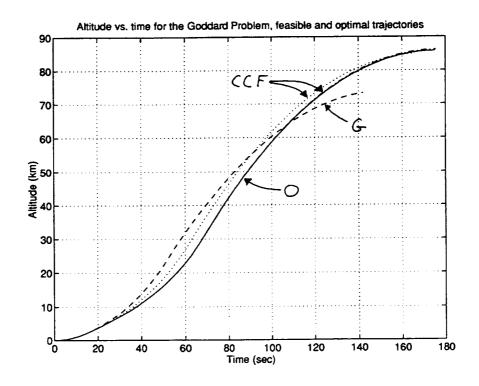
 $dv/dt = \{T - D[\rho(h), v]\}/m - \mu_E/(h+R_E)^2$ 

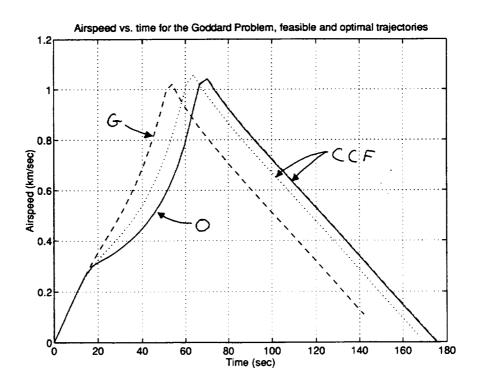
 $dm/dt = -T/v_{exhaust}$ 

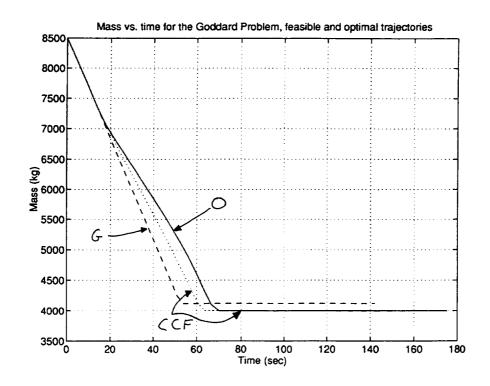
 $0 \le T \le T_{max}$ 

 $m_{\min} \leq m(t_f)$ 









# **Execution Times for Typical Problems**

Problem	N	n	m	RK/	IBM RISC
				$\Delta t$	CPU sec
Max. range, feasibility only	90	4	1	2	498
Max. range	90	4	1	2	10,616
Goddard, very good guess	51	3	1	1	420
Electric-prop., good guess	76	5	2	1	738
Electric-prop., cost- constrained feasibility	76	5	2	1	949

<sup>\*</sup> n increased by 1 to fit into Mayer form

# **In-Class Experience**

Used by 4 students in senior elective at Technion who...

- ... received 3 hours of lecture, a handout, and individual help.
- ... programmed and solved the maximum-range problem.
- ... learned and used cost-constrained feasibility approach to improve guess of solution.
- ... helped find ways to make the code more userfriendly.
- ... are slated to solve the Goddard problem and the minimum-time interception problem.

### **Conclusions**

- Trajectory optimization tool relatively easy to use.
- Tool admits a wide variety of problem forms.
- Tool has robust convergence.
- Execution CPU times range from minutes to hours.

  Dependent on problem dimensions, accuracy of first guess, initial penalty parameter
- Dynamic feasibility and cost-constrained dynamic feasibility problems usually solved more rapidly.
- Best approach first solves sequence of costconstrained feasibility problems.

# Planned Future Work (the LORD willing)

- Faster nonlinear programming convergence.
- Automated cost-constrained-sequence approach.
- More efficient/reliable Hessian and gradient calcs.
- More efficient QP solution procedure.
   low-rank updates
   special dynamic programming structure
- Improved user friendliness, error checking.
- Detailed instruction manual with examples.

	- -	

#### Geometric Considerations in the Solution of **Optimal Control Problems**

K. D. Mease

Mechanical and Aerospace Engineering University of California, Irvine

Workshop on Trajectory Optimization Methods and Applications

AIAA Guidance, Navigation and Control Conference Baltimore, MD August 7-9, 1995

#### Research Overview

#### Objective:

Time-scale decomposition for finite-dim nonlinear dynamical systems

#### Applications:

- Reduced-order models
- Composite control design
- Numerical solution of IVPs
- Numerical solution of BVPs, e.g., for Hamiltonian systems that constitute necess, conds, for optimal control

Collaborative research with S.-H. Lam, Princeton Univ., sponsored by NSF

#### Approaches

- Existing
  - Singular Perturbation Method
  - Limited to standard form
  - Geometric Singular Perturbation Theory (Fenichel)
  - Computational Singular Perturbation (CSP) Method (Lam)
    - Stiff IVPs
- Current Research
  - Geometric Interpretation of CSP for stiff IVPs (IFAC Nonlinear Control Systems Design Symposium 6/95)

  - Extension of CSP to stiff BVPs (e.g. Hamiltonian BVPs of optimal control, topic of this falk)

#### Optimal Control Problem

$$x(t)\in\Re^n,\ u(t)\in\Re^m,\ u(\cdot)\in PWC[0,t_f]$$

$$\min \quad J = \Pi(x(t_f)) + \int_0^t L(x, u) dt$$

subject to 
$$\dot{x} = f(x,u)$$
,  $x(0) = x_0$ 

#### Categories of Solution Methods

Direct: min J

- transform to and solve as NLP problem
- function space gradient
- non-derviative based methods

Indirect: find solution to necess./suff. conds.

#### Hamiltonian Boundary-Value Problem

1st order necess. cond. via maximum principle take form of Hamiltonian BVP

Optimal Hamiltonian:

$$H = L(x, u^*(x, \lambda)) + \lambda f(x, u^*(x, \lambda))$$

Hamiltonian System:

$$\dot{x} = f(x, u^*(x, \lambda)) = H_{\lambda}$$

$$\dot{\lambda} = -H_z$$
  $\lambda(t) \in \Re^n$ , row vector

**Boundary Conditions:** 

$$x(0) = x_0$$

$$\lambda(t_f) = \Pi_x(x(t_f))$$

#### Liouville's Theorem

"Flow" of Hamiltonian system preserves volume i.e., Hamiltonian "fluid" is incompressible

since

$$div\binom{H_{\lambda}}{-H_{x}} = \sum_{i=1}^{n} \frac{\partial^{2} H}{\partial x_{i} \partial \lambda_{i}} - \frac{\partial^{2} H}{\partial \lambda_{i} \partial x_{i}} = 0$$

**Implication** 

Fast decaying behavior ⇔ Fast growing behavior → Potential for Sensitivity

#### Vector Field Splitting

$$\begin{pmatrix} x \\ \lambda \end{pmatrix} = \begin{pmatrix} H_{\lambda} \\ -H_{x} \end{pmatrix}$$

=  $A_{slow} f_{slow} + A_{fast, mable} f_{fast, stable} + A_{fast, unstable} f_{fast, unstable}$ where e.g.  $A_{slow} f_{slow} = a_{s1} f_{s1} + a_{s2} f_{s2} + \dots$ 

and the  $a_{ii}$  are slow basis vectors

Invariance properties:

 $(1) f_{fast,unstable} = 0$  on fast stable manifold

 $(2)f_{fast,stable}=0$ 

 $f_{fast, unstable} = 0$  on slow manifold

 $(3) f_{fost,stable} = 0$  on fast unstable manifold

Furstable Saddle Structure of Fast Stable and Unstable Manifolds along Slow Manifold

#### Conclusions

- Geometric structure of Hamiltonian flow offers insight that suggests indirect numerical solution methods for optimal control
- Viable method for nonlinear regulator problem has been developed
- Extension to two time-scale problems appears feasible and is next step
- Knowledge of geometric structure should also allow simplified (reduced-order) nearoptimal guidance law development

.

# Recent Developments In Trajectory Optimization At McDonnell Douglas

By
R. L. Nelson
Senior Principal Engineer Scientist
McDonnell Douglas Aerospace

#### **Topics**

- SNOPT, a large scale sparse optimizer
  - Main properties, current and future work
  - Timing comparison: OTIS/NZOPT vs. OTIS/SNOPT
- OTIS Development
  - Nodes/SNOPT
  - Tabular Data Smoothing
- Applications

McDonnell Douglas Aerospace

- HSCT/Noise Modeling
- 6 DoF DC-Y rotational landing maneuver video
- AF49, a variational launch vehicle program
  - Program Description/History
  - SEAMLESS: A configuration design tool using linked EXCEL spreadsheets

#### **SNOPT, A Large Scale Sparse Optimizer**

#### Main Properties

McDonnell Douglas Aerospace

- Globally convergent
- As robust as NPSOL and NZOPT

#### Current Activity

- A Limited Memory (LM) version of SNOPT developed
   The current LM algorithm approximates the Hessian of the Lagrangian H by a diagonal matrix plus a fixed number K of rank two matrices which need never be summed, since only the product of H with a vector is needed in the QP subproblem
- Initial version chosen for speed of implementation and simplicity of coding

#### Future Work

- More complicated LM formulas are known that require half as many multiplications
- These methods will be implemented next

#### Timing Comparisons: OTIS/NZOPT vs. OTIS/SNOPT

#### Problem

McDonnell Douglas Aerospace

Minimum time to climb with two phases and a nonlinear dynamic pressure constraint

- SPARC station model 20/61 (128 MB memory) with optimized FORTRAN
- Comparison (in seconds)

NODES	<u>VARIABLES</u>	OTIS/NZOPT	OTIS/SNOPT
20 = (10, 10)	184	49.1	14.9
30 = (15, 15)	274	163.8	27.2
40 = (20, 20)	384	449.4	39.0
50 = (25, 25)	454	1368.3	59.0
60 = (30, 30)	544	2194.0	92.5
70 = (35, 35)	634	4324.5	122.1
80 = (40, 40)	724	6440.3	169.9
90 = (45, 45)	814	9348.0	203.4
100 = (50, 50)	904	14060.5	249.8

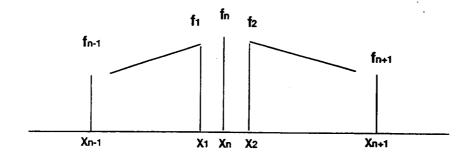
#### Nodes/SNOPT

- McDonnell Douglas Aerospace
  - User/Nodes Issue
  - Node Refinement Descriptions
    - Solve a sequence of optimization problems with increasing number of nodes
  - Warm Start Procedure Developed for SNOPT
    - At the end of an optimization run w.r.t. a set of nodes a new procedure that determines where additional nodes should be placed is in development
    - This new procedure will allow the solution of problems with a refined nodal structure to be reached in relatively few iterations

#### **Tabular Data Smoothing**

McDonnell Douglas Aerospace

- Technique developed that eliminates the need to smooth tabular data by the user
- Uses user tabular data "as is"
- Smoothing algorithm as a Linear/Quintic Patch Technique



- Eliminates possible large wiggles due to the quintic fits
- Univariate, bivariate completed / trivariate in development

#### **HSCT Flight Path Optimization with OTIS**

- Minimize Fuel Consumption
   Find Optimum Attitude and Throttle Control Time Histories
   Satisfy Constraints
  - 4% Minimum Climb Gradient,  $h \le 10K$  ft ( $\gamma \ge 2.3^{\circ}$ )
  - 4000 ft/min Max Climb Rate ( hmax)
  - Angle of Attack, 0 ≤ AOA ≤ 13 deg
  - Attitude rate, |θ|≤4 deg/sec
  - Throttle Factor, 0.2 ≤ eta ≤ 1.0
  - Velocity, V,

McDonnell Douglas Aerospace

 $\{h \le 10 \text{ Kft: } V \le 250 \text{ KCAS}\},$ 

 $\{10 \text{ Kft} \le h \le 25 \text{ Kft: } V \le 400 \text{ KCAS}\},$ 

 $\{h > 25 \text{ Kft: Mach} \le 0.95\},$ 

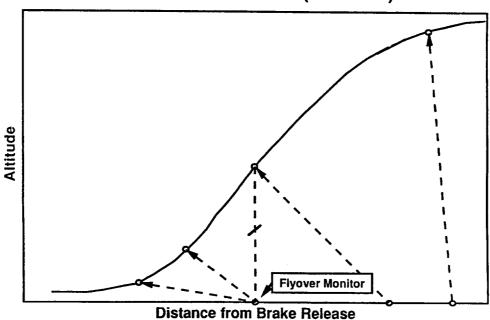
where h = altitude;

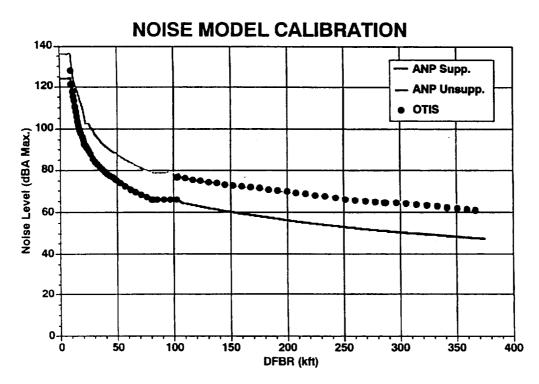
- V ≥ Vo
- Fly over noise monitor ≤ 89 dBA
- Community noise ≤ [89-7xlog(D/DO)/log2] dBA, where D=ground-track DFBR and DO=ground-track DFBR to flyover monitor
- Sideline noise ≤ 92 dBA

#### **BUILD NOISE MODEL**

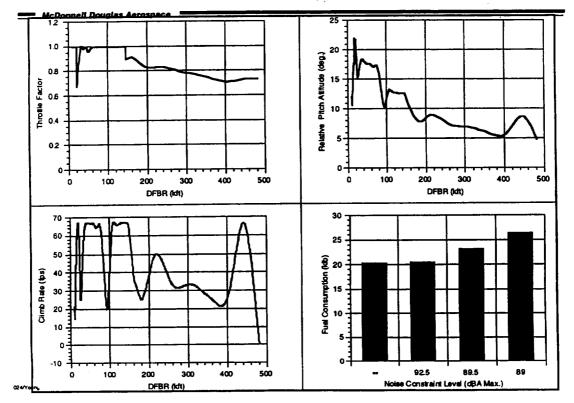
OTIS/ANP NOISE MODEL	ANOPP METHOD
Tabulate a noise versus power and distance (NPD) map by integrating time histories of levelflight, constant speed and thrust flyovers	<ul> <li>Calculate trajectory thrust, position, and velocity histories.</li> </ul>
At each point along trajectory, interpolate the NPD tabulation to determine the noise level at the ground point at minimum distance to the flight path	<ul> <li>Calculate noise level at specified ground locations by integration of the noise time histories over the entire flight path.</li> </ul>

# **BUILD NOISE MODEL (CONT'D.)**





HSCT Configuration 8A: Min. Fuel w/Mach at 31K ft > 0.9 and Noise < 89.5 dBA Max. TFBR to 31 Kft = 736.0 sec;  $\Delta$ Wf = 23,200 lb



#### **Application: DC-Y Rotational Maneuver Video**

- Problem: Soft land the DC-Y vertically on the ground with minimum fuel expended
- 6 DoF equations of motion
- Control: Thrust magnitudes and gimbal angles of each engine (4)
- Flight Profile

McDonnell Douglas Aerospace

- Vehicle starts at about 2800 ft above ground with nose pointed down
- To pitch the vehicle's nose upward (vehicle is aerodynamically unstable), an aero moment is generated by stowing flaps (not shown on video)
- The control variables are optimized, which results in differential throttling of the engines, in order to vertically land the vehicle
- The body attitude rates as well as all three components of velocity must be zero when the vehicle lands
- Results: The OTIS optimization resulted in a 53% propellant reduction from the previous best known result
- Video Tape: There are a number of runs on the tape. The tape shows the landing in real time and one-tenth real time from different perspectives of the vehicle

#### AF49, A Variational Launch Vehicle Program

McDonnell Douglas Aerospace

- Program Description/History
  - Developed in the 60's
  - Primarily used for optimization and sizing of launch vehicles by the Delta Launch Vehicle Program for over 30 years
  - Uses a variational optimal control (angle-of-attack and/or thrust) program, with moment balance
  - Solves a second order Taylor series expansion of the boundary conditions as a function of the initial control and control rates
  - Varies vehicle gross weight and resolves the TPBVP to max weight to orbit
  - Total process takes about 30 sec. on an IBM 486 50 Hz PC

# AF49, A Variational Launch Vehicle Program

- McDonnell Douglas Aerospace
  - SEAMLESS: A vehicle design trade tool
    - Linked EXCEL spreadsheets that call each other
    - Engineering disciplines turned into spreadsheet databases and calculations: Aero, Prop, Geom, Loads, Structures, Cost, Performance, etc.
    - AF49, the Performance discipline, compiled in Fortran and converted to a DLL for EXCEL. AF49 is a function call from EXCEL. Returns ΔV's back to SEAMLESS.
    - Performs perturbations about a given design
    - Generates minimum cost design trades

# DAB ASSOCIATE RECEIVED WITH PROMINED

# Presented to the Workshop on Trajectory Optimization Methods and Applications

August 7, 1995

DAB Engineering, Inc. 2155 S. Valley Hwy., Ste 201 Denver, CO 80222 (303) 757-6425 Voice (303) 757-1215 FAX dbaker@dab.com

# Overview of DAB Ascention Windows

- DAB Ascent for Windows (DAW) is a 3-DOF launch vehicle and reentry vehicle simulation program
- DAW is a purely commercial product
  - No government funding has been used for any part of its development or development of its predecessor (DAB Ascent for DOS)
- · Vehicle Database of all major launch vehicles also available
- DAB Ascent for Windows has been converted for 32 bit OS's
  - Ready for Windows95
  - Already operational on Windows NT
  - Win32S allows Windows 3.1, WFW3.11 to run 32 bit applications
  - Currently shipping version 1.2.2 which is a 32 bit application

# Which is Using Date Assegni

• Lockheed Martin (four different sites)

(Partial list)

- The Aerospace Corporation (CA & FL)
- Space Systems/Loral
- Pratt & Whitney
- Akjuit Aerospace (Space Port Canada)
- Bristol Aerospace
- AeroAstro
- Kistler Aerospace
- Analex
- ComSat
- Allied Signal Aerospace
- Eutelsat
- ACTA, Inc.
- Naval Research Laboratory
- MCHI (Ellipsat)

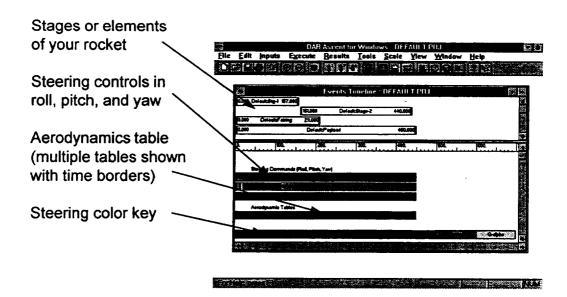
# BANG Garabilla Cyantau

- 3 Degree of freedom rocket flight simulation
- Forces:
  - Thrust with atmospheric engine pressure losses
  - Gravity up to J₄
  - Aerodynamics (Cd, Cl, Cy, Cd/alpha total, Cl/alpha, Cy/beta)
  - Buoyancy (displaced air volume times air's density)
- User defined central body, atmosphere, and winds
- Seven steering options (by roll, pitch, and/or yaw)
  - Hold to pad
  - Follow Vr, Vr with winds, or Vi vectors
    - » User defined offset angles from vectors
  - Inertial turn/user defined
  - Inertial turn/computer controlled to meet target
  - Constant Q-alpha
- Optional Vehicle Database with over 40 of the worlds launcher

# Daiming the Vehicle

- The following four slides will cover:
  - Sequence of flight events
  - Thrust, flow rate, weight, etc.
  - Aerodynamics
  - Steering
- · All input data is entered and edited via dialog boxes
- Your entire simulation is saved as one project file (.PRJ)
- Double clicking bars on the Event Timeline opens the edit dialog boxes

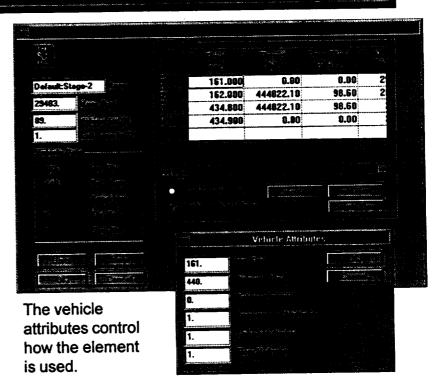
# Sequence of High Events



# There from Pairs Wising the

By doubleclicking on any element bar (rocket stage), you can open this dialog box to edit key parameters of this rocket element.

This defines a stage, independently of how the vehicle will use it.



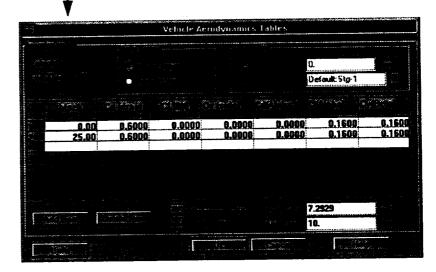




Aerodynamics are entered via this dialog box.

You can have any number of complete tables, one for each aerodynamic configuration of the rocket.

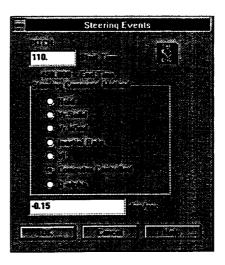
Each table can have any number of entries by Mach number



# Steering

Each steering bar can be double clicked to get to this dialog box.

Here you change the type of steering control, when it starts, and what its parameter is.



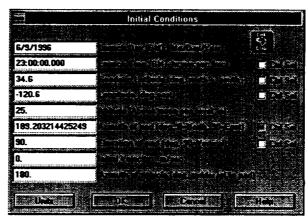
# Inital Launeh Conditions

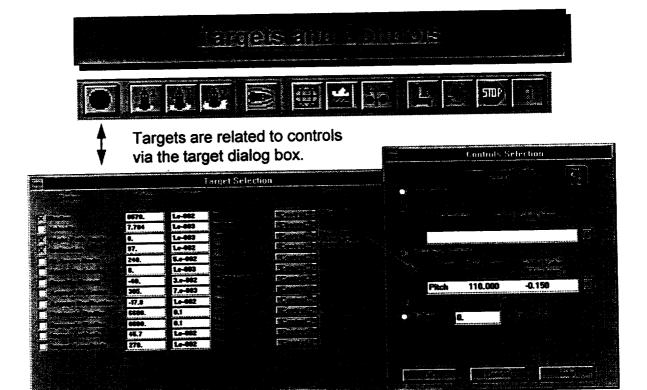




The initial conditions are defined with this dialog box.

You can open this dialog box directly from the icon bar across the top of the main window.







#### • Central body (planet)

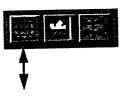
 This dialog box controls the values associated with the central body

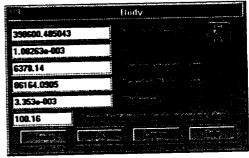
#### Atmosphere

 The atmosphere is NASA's 1976 std.
 You can build any tabular atmosphere for any planet or moon.

#### • Winds

 You can input any wind profile including sheers and twisting winds with altitude.





# Running the Simulation

- · One iteration: 1
  - Flies the rocket on the reference trajectory without changing any controls
  - Use this option for initial guesses for the proper controls
- · Target only: T
  - Uses the assigned controls from the target/control dialog box to meet the desired targets
  - Free controls are not varied
- · Target and optimize the trajectory: O
  - Target controls are used along with the free controls to find the optimal targeted trajectory
- Abort and Pause



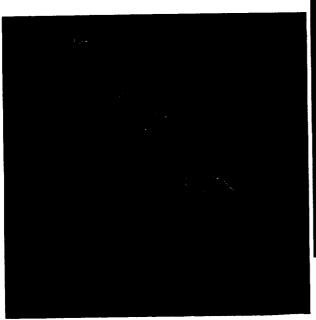
Note: Running a simulation is only possible if you have purchased DAB Ascent for Windows

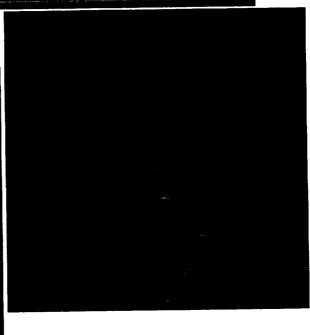
## Office Products

- The following five slides will show you the output products of DAB Ascent for Windows
- DAB Ascent for Windows produces the following output products to help you visualize your simulation
  - Mercator and 3-D perspective view
  - X-Y plots
  - Tabular listings
  - Reports
  - Flight and burnout conditions

# Westerson and Sad Paradediction

These plots were simply copied into PowerPoint via Ctrl-C and Ctrl-V

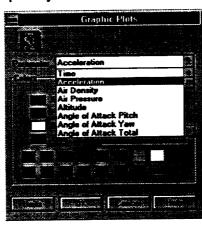


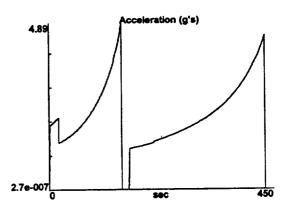




Plots can be made of virtually any flight parameter

Use the Graph pick list to select which plots you wish to see





This plot was cut and pasted directly into PowerPoint

8/4/95

**DAB-16** 

# Tabular Listings

User defined tabular entries

User defined units (SI or English)

User defined start and end times

Full scrolling available during run-time

3	Tabula	r Listing	<b>5</b> 7	
Time	Accel	Vr	Altitude	
(sec)	(g's)	(km/sec)	(lem)	
0	7.064e-007	1.031e-017	0.02272	
0.1	7.064e-007	1.745e-017	0.02272	
0.5	7.064e-007	9.314e-018	0.02272	
2.167	1.884	0.01429	0.03458	
3.833	1.906	0.02894	0.07055	
5.5	1.928	0.04394	0.1312	
7.167	1.949	0.0593	0.2172	8
8.833	1.971	0.07501	0.3291	
10	1.986	0.08622	0.4231	
11.67	2.008	0.1025	0.5804	
13.33	2.029	0.1192	0.765	30
15	2.052	0.1361	0.9776	
16.67	2.074	0.1534	1.218	
18.33	2.097	0.1711	1.488	
20	2.121	0.1891	1.786	6
20.1	1.384	0.1898	1.804	
21.1	1.385	0.1937	1.994	
22.1	1.402	0.1976	2.188	
23.1	1.4	0.2016	2.385	

# Regers

Many different reports are available

A mass report is shown in the lower right of this slide

Other reports available are:

Velocity losses Staging Burnout conditions Flight maximums Final vehicle state

2			Mass Repo			
			MASS REPOI mass in kg			
Event	Time	Total	Stage-1	Stage-2	FAIRING	PAYLOAD
Initial	0.0	159748.0	122015.0	29483.0	750.0	7500.0
Prop Exp		-117950.4	-117950.4			
Released		-4064.6	-4064.6			
Subtotal	157.0	37733.0	0.0	29463.0	750.0	7500.0
Prop Exp		-4880.7		-4880.7	0.0	
Released		-750.0			-750.0	
Subtotal	211.0	32102.3	0.0	24602.3	0.0	7500.0

# Nonce figure in the

Burno	ut Conditio	ns 🏻 🚍
Time (sec)		450.00
Mass (kg)		7500.00
Altitude (kn	n)	202.93
Vi (km/sec)	1	7.7666
FPA i (deg)		-0.0000
i (deg)		96.9988
RAAN (deg	)	118.845
Radius (kn	n)	6577.99
Rp (km)		6518.37
Ra (km)		6577.99
Target	Error	Tol
Radius	-6.7 <b>e</b> -003	1.0e-002
FPA i	-4.4e-006	1.0 <del>c-</del> 003
i	-1.2e-003	1.0 <del>c-</del> 002

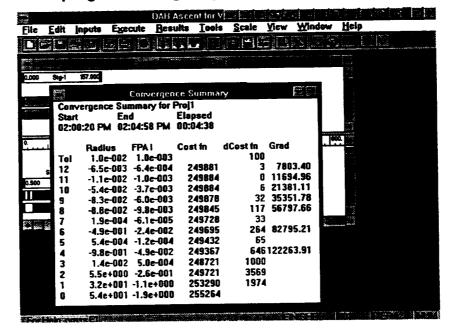
Flight Conditi	ions <b>2</b>
Time (sec)	450.00
Alt (km)	202.93
Vr (km/s)	7.8374
Vi (km/s)	7.7666
Accel (g's)	0.000
Mach	13.78
Q (kPa)	0.01
Drag (N)	0.00
Lift (N)	0.00
Side Force (N)	0.00
Thrust (N)	0.00
Mass (kg)	7500.00
Alpha (deg)	-4.722
Beta (deg)	1.444
FPAr (deg)	-0.0000
Pitch (deg)	-4.64

Flight conditions allow you to see the progress of your simulation while it proceeds

Burnout conditions include target errors so you can watch the progress of the targeting process

# Montos Conversens

Shows progress of targeting and optimization iterations



# Vehice Database inventory

- Database currently consists of the following vehicles
- Data collected from public sources, fidelity varies among vehicles

#### Domestic Vehicles

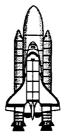
- Atlas 2, 2A, 2AS
- Delta 6920, 6925
- Delta 7930, 7920, 7925
- Scout, Scout-II
- Space Shuttle
- Titan-II, III, IV
- Titan-IV / Centaur

#### Micro-Launchers

- •Conestoga-1229, 1620
- •LLV1, LLV2
- ·Pegasus, Pegasus-XL
- •Taurus, Taurus-XL
- ·MSLS
- •X-34A

### International Vehicles

- Ariane 40, 42P, 42L, 44P, 44L, 44LP
- Ariane 5
- ° Energia
- ° Energia-M
- H-II
- ° Long March 2E/HO, 3, 3A
- ° Kosmos
- Soyuz
- ° Tsyklon-2, 3 (Cyclone)
- Proton
- ° Zenit-2.3



# Company History

- DAB Engineering was incorporated in 1987 and has been active since 10/92
- · David Baker is founder and President
- Roger Kovacs is V.P. of product development
- First sale was made in 2/93
- Over 50 DAB Ascents have been sold to date
- DAB Engineering also provides contracting/consulting services and has been under contract with the following companies
  - CTA, CSMC, ATA, USAFA, Cybersystems, CU/Space Grant, Lockheed-Martin, and Akjuit Aerospace

# Some Recent Developments in Computational Optimal Control

Renjith Kumar & Hans Seywald

Analytical Mechanics Associates, Inc. 17 Research Drive Hampton, VA 23666

> Phone: (804)865-0944 Fax: (804)865-1881 e-mail: ama@infi.net

> > AMA Inc.

### **Outline**

- A Concatenated Approach to Trajectory Optimization (CATO)
  - Principle of Optimality
  - The Algorithm
  - Numerical Example(s)
- A Receding Horizon Nonlinear Feedback Approach
  - Gut Feeling
  - The Algorithm
  - Numerical Example(s)

### **Principle of Optimality**

• For optimal control problems in Mayer form the Principle of Optimality states:

"If a trajectory is optimal on  $[t_0, t_f]$ , then each sub-arc  $[t_1, t_f]$  of the trajectory optimizes the same original cost criterion."

• The method Concatenated Approach to Trajectory Optimization (CATO) is based on the observation that for certain classes of optimal control problems (e.g., automomous minimum time, minimum energy) the Principle of Optimality can be stated in the following form:

"If a trajectory is optimal on  $[t_0, t_f]$ , then each sub-arc  $[t_1, t_2]$  of the trajectory optimizes the same original cost criterion."

#### Note:

For problems of the second type, the total performance of a sub-optimal trajectory will be improved if the performance is improved on a sub-arc of the trajectory, while the remainder of the trajectory is kept unchanged.

– AMA In

# Concatenated Approach to Trajectory Optimization (CATO)

The basic idea is to improve a given (non-optimal) starting trajectory by iteratively optimizing short, partly overlapping subarcs.

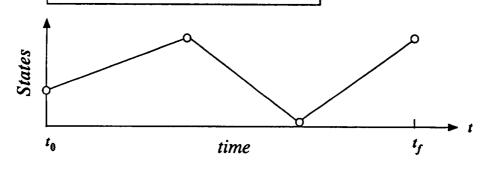


Instead of solving one "big" problem, one has to solve many "small" problems now. This can greatly reduce the CPU time and storage area requirements for a direct optimization approach.

# The CATO Algorithm: Step 1

- Consider a minimum-time problem
- Let the computing power be limited to a 4 node discretization
- Let the desired solution be a 13 node solution

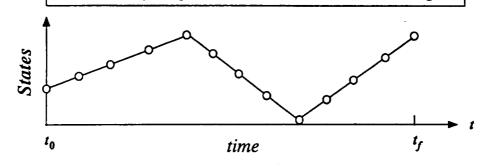
Step 1: Solve the 4-node discretized problem

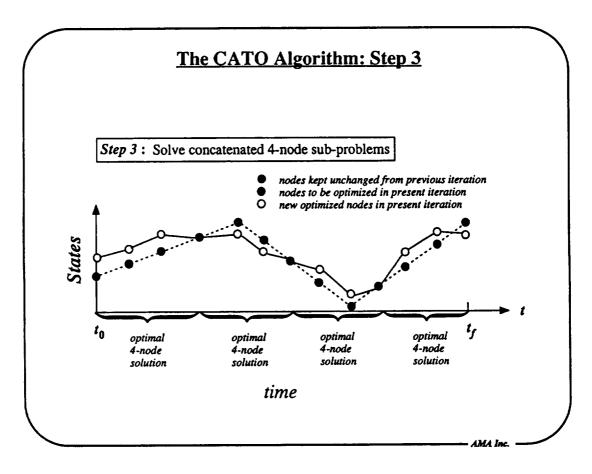


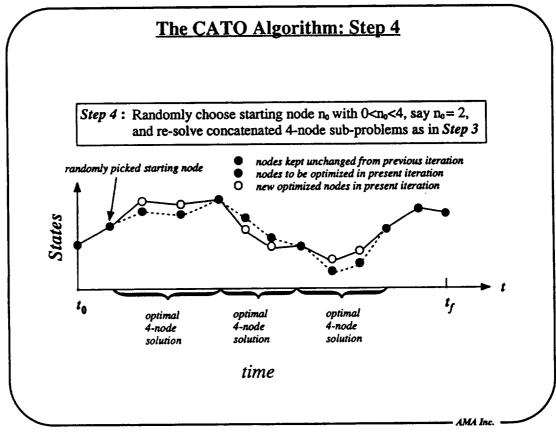
AMA Inc.

### The CATO Algorithm: Step 2

Step 2: Linearly interpolate 4-node solution and obtain 13 node guess







### The CATO Algorithm: Step 5

Step 5: Repeat Step 4 iteratively until convergence is achieved

AMA Inc.

### **Remarks**

- The main advantage of this technique is the reduction in computing power requirements. The method is also well suited for parallel computing.
- The class of problems that can be solved with this method includes autonomous minimum—time problems and minimum energy problems.
- The extension of the CATO approach to the most general optimal control problem is a current research topic!!
- The obtained trajectory satisfies all physical constraints even before overall convergence is achieved

# Numerical Example: Minimum Time To Climb of a Fighter Aircraft

Cost

 $\min t_f$ 

# **Differential Constraints and Control Constraints**

$$\dot{E} = (\eta T - D) \frac{v}{mg}$$

$$\dot{h} = v \sin \gamma$$

$$0 \le \eta \le 1$$

$$\dot{\gamma} = (n - \cos\gamma) \frac{g}{v}$$

$$-10 \le n \le 10$$

# **Boundary Conditions**

$$E(0) = 2668m$$

$$E(t_f) = 38029.207m$$

$$h(0)=5m$$

$$h(t_f) = 12119.324m$$

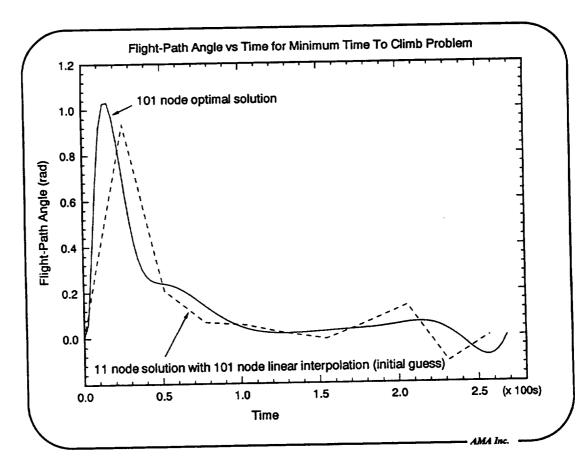
$$\gamma(0) = 0$$
rad

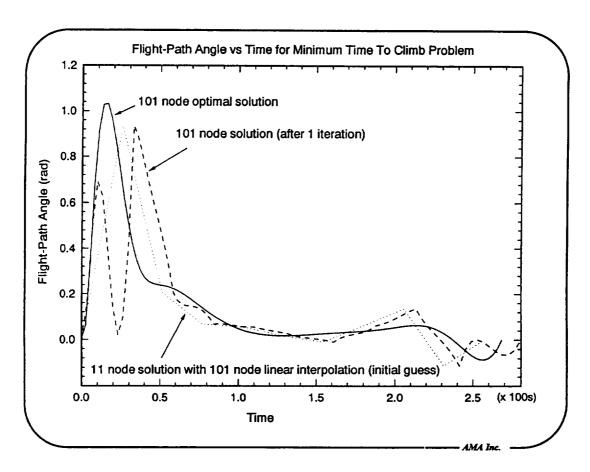
$$\gamma(t_f) = 0$$
rad

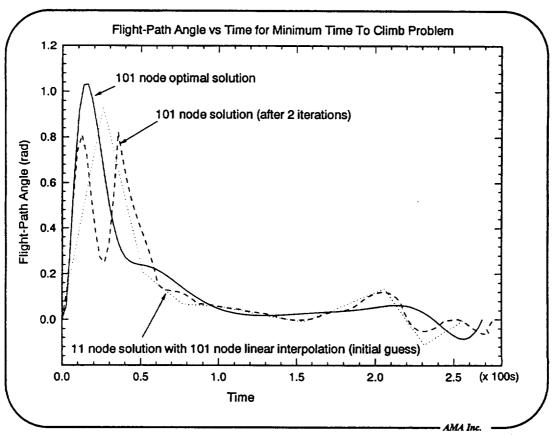


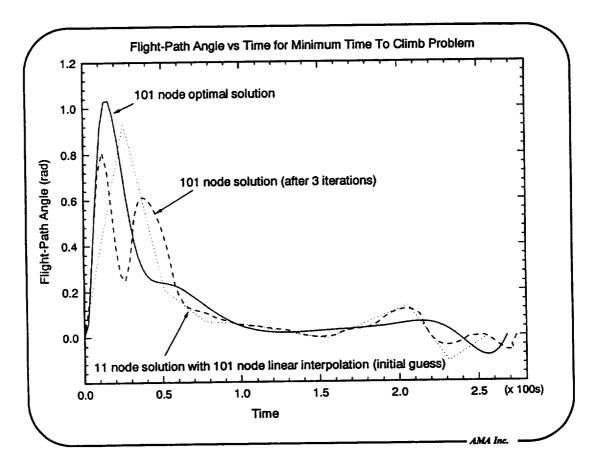
Initial Conditions: Shortly after take-off

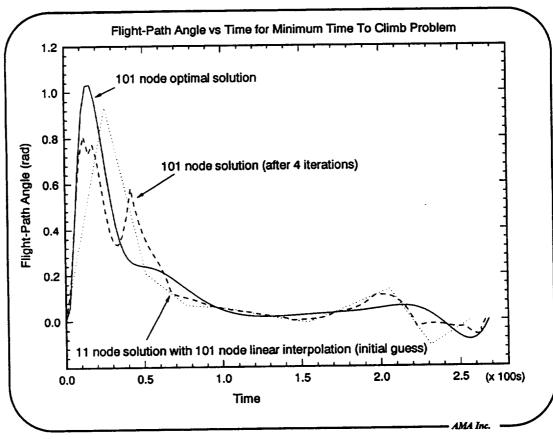
Final Conditions: Dash point

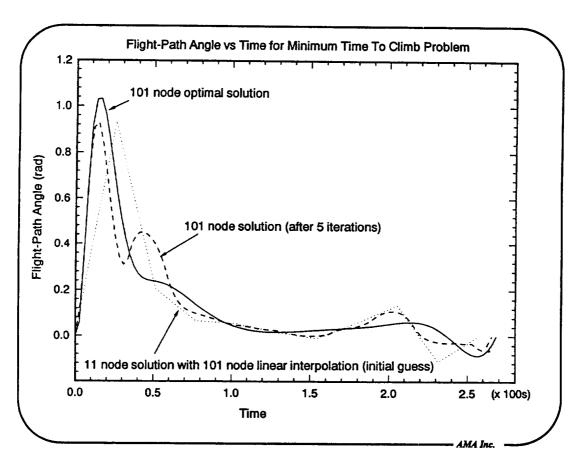


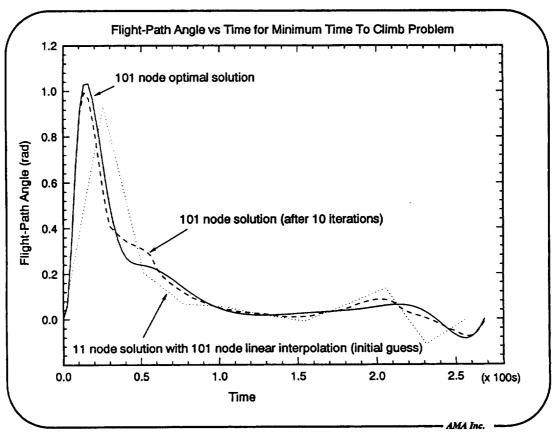


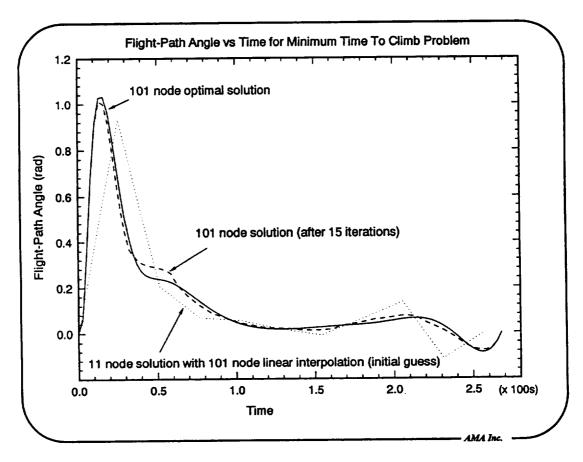


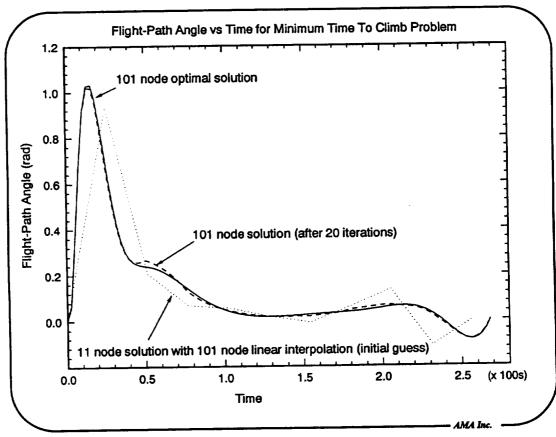


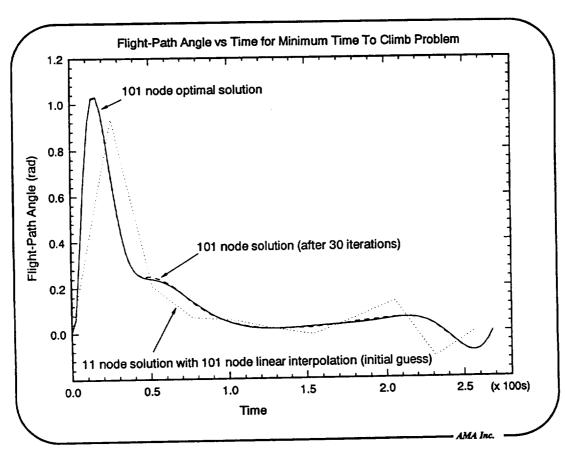


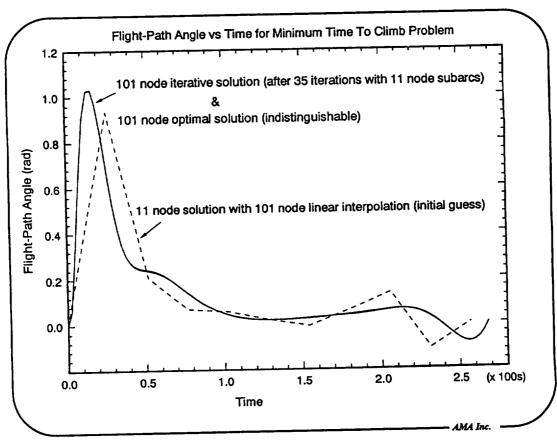


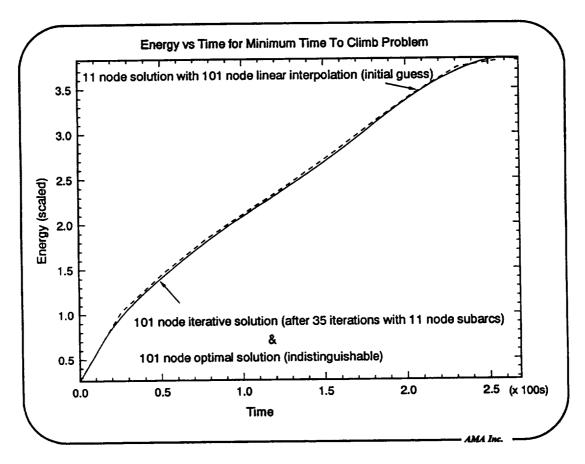


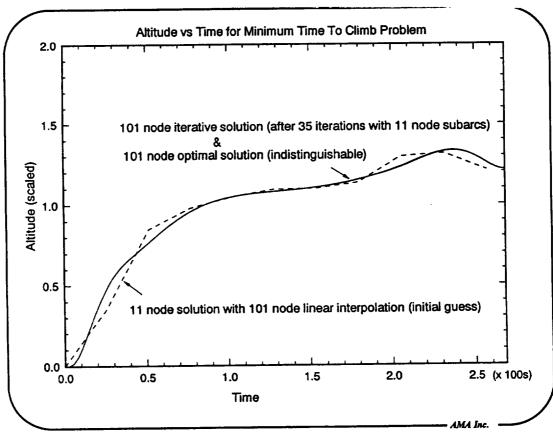












### **Other Examples**

• Following examples were also solved using this technique -

Flexible Body Control (2 carts and spring)

Fuller Problem (second order singular arc, chattering junctions)

Zermelo Problem (river crossing problem)

- AMA Inc

### Receding Horizon Nonlinear Feedback Guidance

• Gut feeling

If you are driving to LA from NY and the objective is to minimize fuel or time, would you really need to make instantaneous control actions to account for every curve and pot-hole miles away? Do you improve your performance index a lot by doing this?

### Answer would be NO!

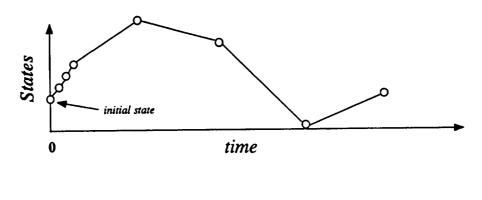
 TODI captures the general dynamics of the problem with little nodes (as seen for many examples).

So, why not discretize with nodes densely placed at current time to capture immediate dynamics very well, and use sparsely placed nodes for the rest of the future trajectory to capture the general dynamics. This idea can be used for optimization and extended to real-time optimal feedback guidance.

### Feedback Algorithm: Step 1

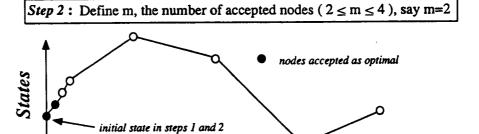
• Let the speed and computing power limit the discretization of the problem to 8 nodes

Step 1: Place 4 dense nodes close to initial time (define node density)
Place 4 sparse nodes for rest of trajectory
Solve the 8 node finite dimensional optimal control problem



AMA Inc.

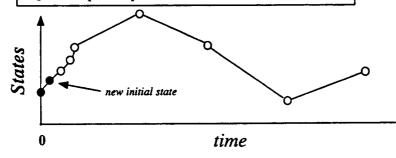
# Feedback Algorithm: Step 2



time

## Feedback Algorithm: Step 3

Step 3: Repeat steps 1 and 2 for most recent initial state



### **Numerical Example:** Minimum Time To Climb of a Fighter Aircraft

Cost

 $\min t_f$ 

### **Differential Constraints and Control Constraints**

$$\dot{E} = (\eta T - D) \frac{v}{mg}$$

$$\dot{h} = v \sin \gamma$$

$$0 \le \eta \le 1$$

$$\dot{\gamma} = (n - \cos\gamma)\frac{g}{v}$$

$$-10 \le n \le 10$$

### **Boundary Conditions**

$$E(0)=2668m$$

$$E(t_f) = 38029.207m$$

$$h(0)=5m$$

$$h(t_f) = 12119.324m$$

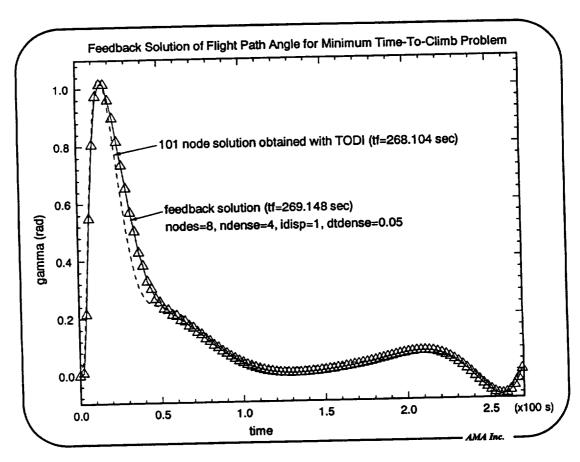
$$\gamma(0) = 0$$
rad

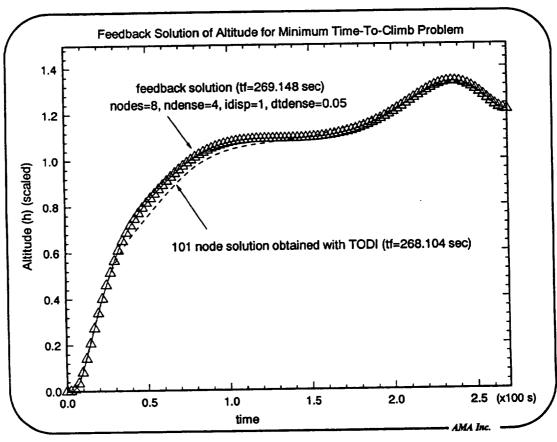
$$\gamma(t_f) = 0$$
rad

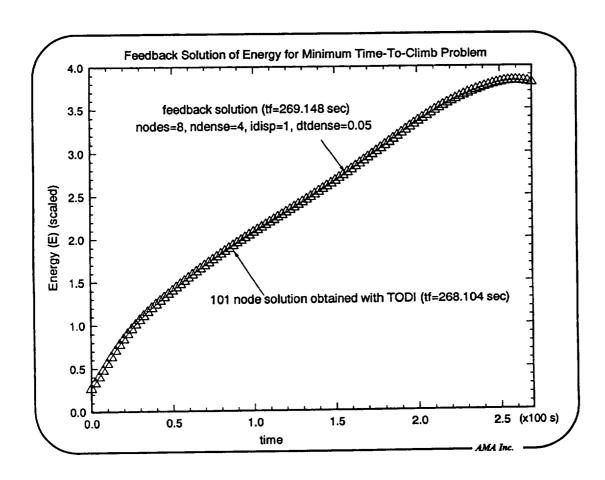


Initial Conditions: Shortly after take-off

Final Conditions: Dash point







## **Other Examples**

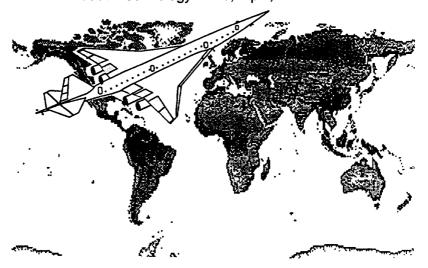
• Following examples were successfully tested using this technique Vertical Rocket Landing Maneuver
Nonlinear Spacecraft Attitude Controller & Momentum Manager

A Differential Game (Pursuit-Evasion)

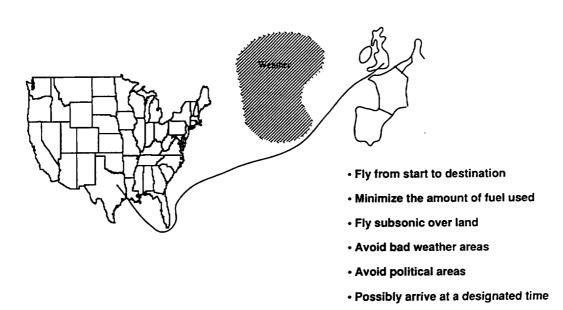
# TRAJECTORY OPTIMIZATION FOR THE HIGH SPEED CIVIL TRANSPORT

Presented at the AIAA Conference Baltimore Md August 7-9,1995

By: R.L. Schultz, D.A. Shaner & M.J. Hoffman Honeywell Technology Center 3660 Technology Drive, Mpls,Mn 55418



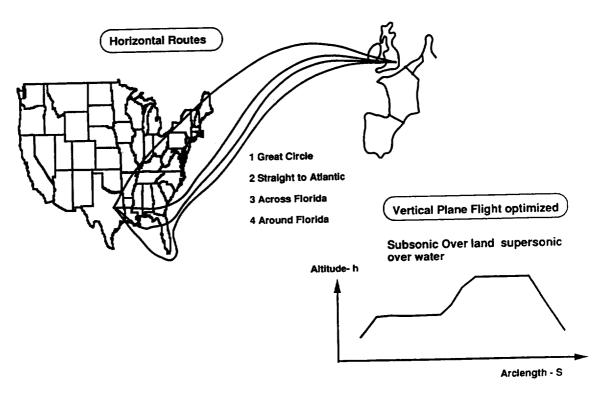
### **HSCT OPTIMIZATION PROBLEM**



• Do not exceed noise restrictions at airports

· Handle emergy abort situations

# **COMPARISON OF FOUR ROUTES - FUEL USED**



# VERTICAL PLANE - ENERGY STATE CALCULUS OF VARIATIONS OPTIMIZATION

#### OPTIMIZATION PROBLEM

$$\begin{aligned} & \min_{h,\pi} \int_{t_e}^{t_f} \sigma T dt \\ & \dot{E} = [T(h,V,\pi) - D(h,V,\alpha)] \frac{V}{m} \\ & \dot{S} = V \\ & \dot{m}_f = \sigma T \end{aligned}$$

#### Constraints

$$E = \frac{1}{2} + gh$$

$$L=W$$

$$M \le 1 \quad \text{over land}$$

$$M \le M_{lim} \quad \text{Supersonic}$$

# SOLUTION FOR OPTIMAL CONTROLS

· CLIMB

$$\min_{h,\pi} H_a \qquad ; \qquad H_a = \frac{O \cdot H_a}{(T - D)^n}$$

$$L=W$$

$$T - D \ge 0$$

$$M \le 1 \quad \text{over land}$$

$$M \le M_{lim} \quad \text{Supersonic}$$

$$\cdot \text{DESCENT}$$

$$\max_{h,\pi} H_a$$

$$L=W$$

$$T - D \le 0$$

• CLIMB / DESCENT SWITCH 
$$\begin{aligned} \min_{h,\pi} H_a &= \max_{h,\pi} H_a \\ h.\pi \end{aligned}$$

M ≤ M<sub>lim</sub> Supersonic

M≤1 over land

### · VALUES FO C

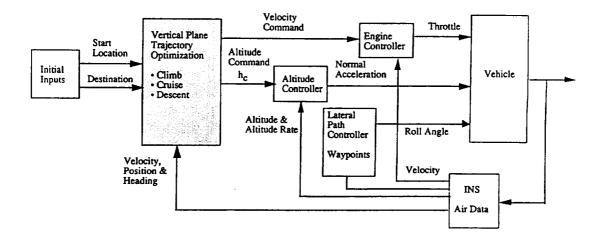
$$C_0 \ge \min_{h,E,\pi} (\frac{\sigma T}{V})$$
; T=D, L=V

### **SOLUTION PROCEEDURE**

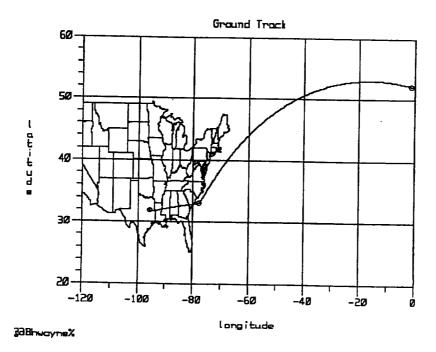
- Pick Co
- Start with climb solution
- Switch when min Ha = Max Ha
- Use descent solution
- · Stop at final energy
- Check if distance is met
- · Pick new value of Co

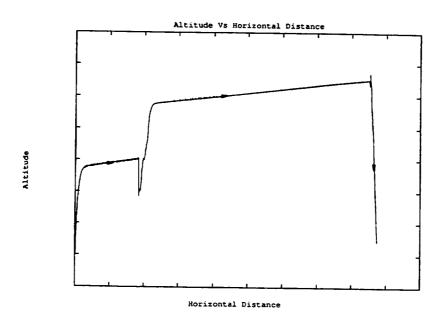
### **ADVANTAGES OF METHOD**

- Compares well with full state gradient method solutions
- Provides analytical functions & insight which can be used in FMS design
- · Low computation time
- Low computation time is impotant when extending to lateral path optimization

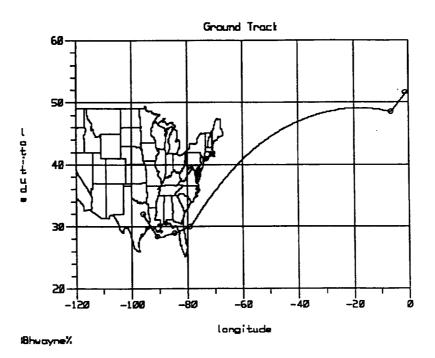


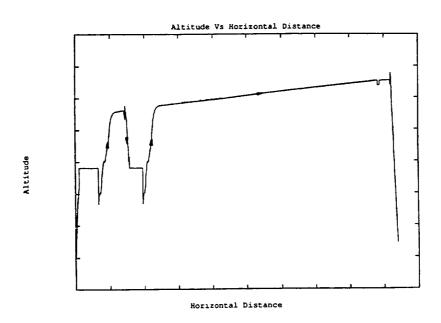
### **Direct to Atlantic**

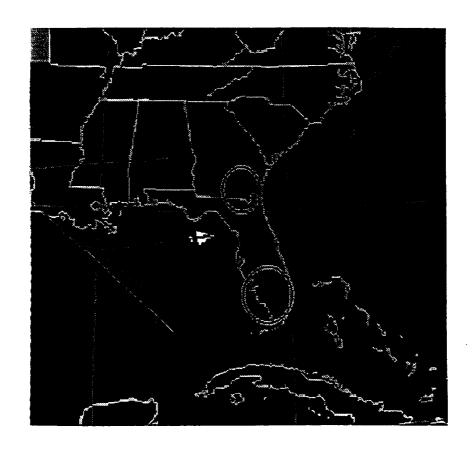


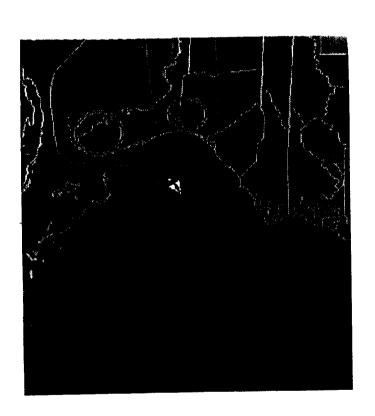


### **Across Florida**

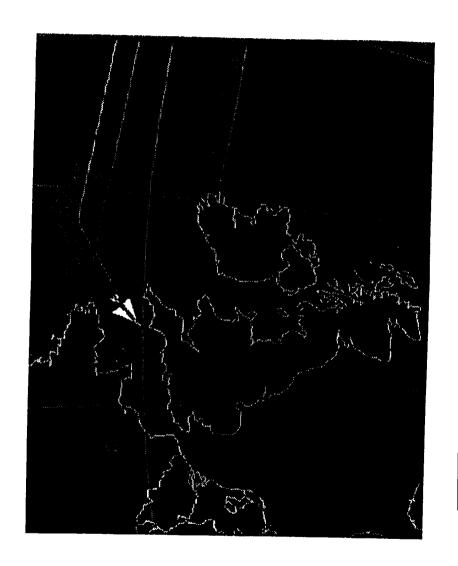




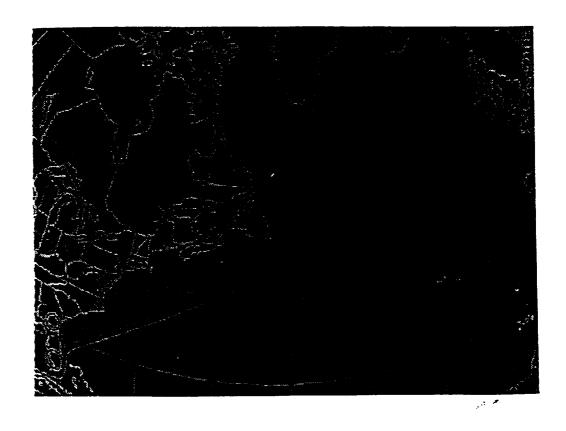








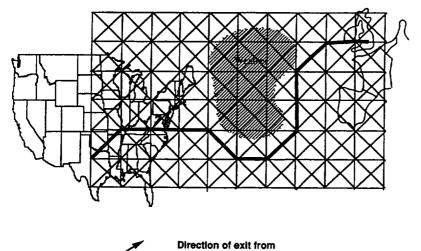




# **ROUTING - COMPARISONS OF RESULTS**

Route	Route Type	Fuel Used	Time
Dallas to London	Great Circle - Supersonic over land 4172 nm	1	1
	Great Circle - Subsonic over land	1.13	1.81
	East to Atlantic 4403 nm	1.10	1.45
	Across Florida 4735 nm	1.15	1.29
	Around Florida 4990 nm	1.19	1.29

# PROPOSED METHOD FOR ROUTE OPTIMIZATION



· Similar method used on route planning for threat avoidance

point depends on

entry direction

# Combined Dynamic Programming & Calculus of Variation

- Decouple vertical & horizontal
- · Construct state space grid
- · Start at end point
- Take horizontal steps in all possible directions to next pointsfirst stage
- Optimize vertical path to next grid point via calculus of variations
- Compute accumulated fuel
   time cost to next point
- If point is reached by more than one path choose lowest cost path
- Process next stage until start point is reached & is the lowest cost point

# TRAJECTORY OPTIMIZATION FOR A SAM USING A MULTI-TIER APPROACH

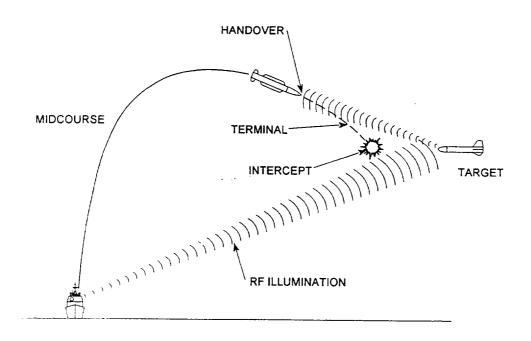
### **CRAIG PHILLIPS**

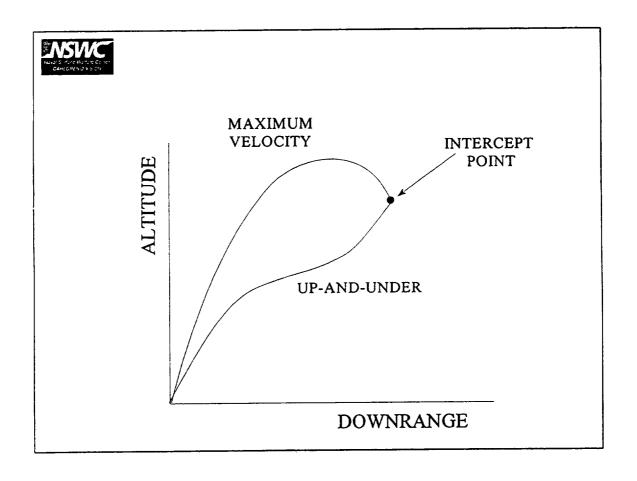
### JUNE DRAKE

### NAVAL SURFACE WARFARE CENTER



### MIDCOURSE/SEMIACTIVE TERMINAL HOMING MISSION





MTOP
MULTI-TIER OPTIMIZATION PROCESS

TRADITIONAL "SIMPLE" MODELS FOR MIDCOURSE
SIMPLE MODEL MATED WITH OPTIMIZER
SOLUTION UNDER VARIABLE GLOBAL CONSTRAINT

MTOP ENVIRONMENT MATES TRADITIONAL APPROACH WITH COMPLEX MOE MODELS

GLOBAL CONSTRAINT VARIABLE IS THE KEY

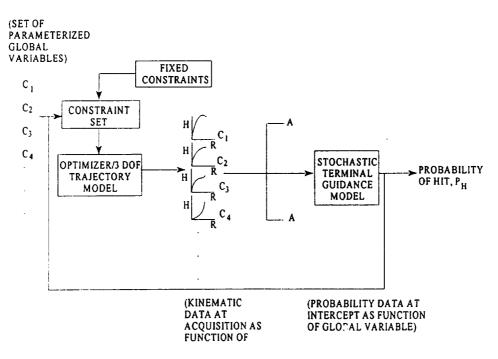
# MOE CANNOT BE BLENDED WITH OPTIMIZER

## DEVELOPED MULTI-TIER PROCESS

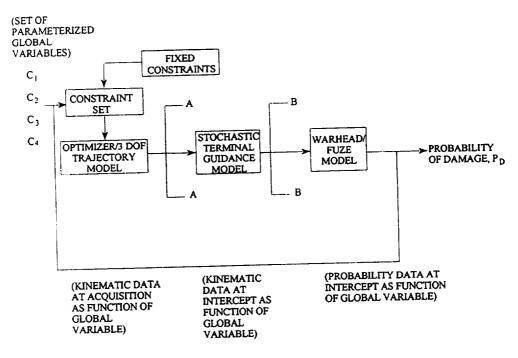
# ALLOWS COMPLEX TERMINAL HOMING MODEL

# SIMILAR TO DYNAMIC PROGRAMMING









## EXAMPLE PROBLEM

# **DESIGN MIDCOURSE TRAJECTORIES**

MOE: PROB. OF HIT

INCLUDE:

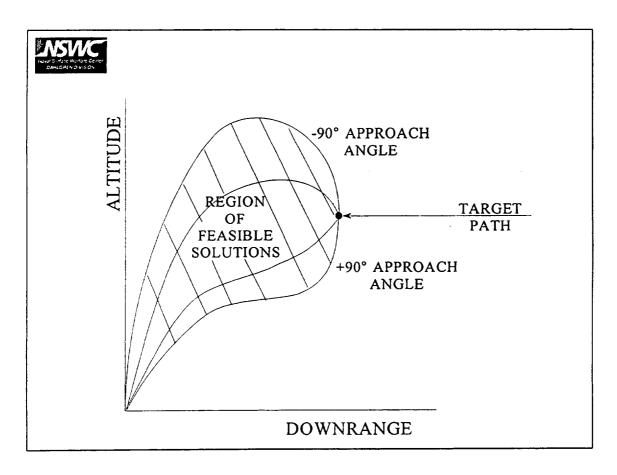
- (1) RADAR NOISE EFFECTS
- (2) RANDOM EVASIVE MANEUVERS
- (3) AIRFRAME RESPONSE

### **MOE**

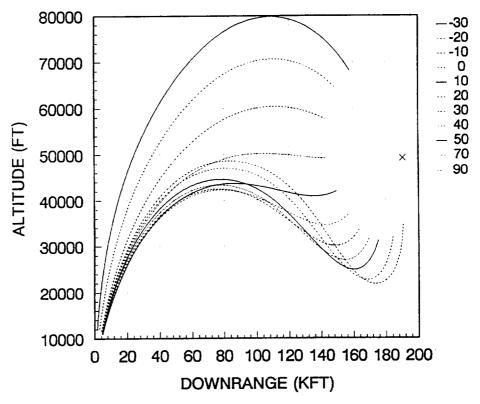
### AVOID "RULES OF THUMB"

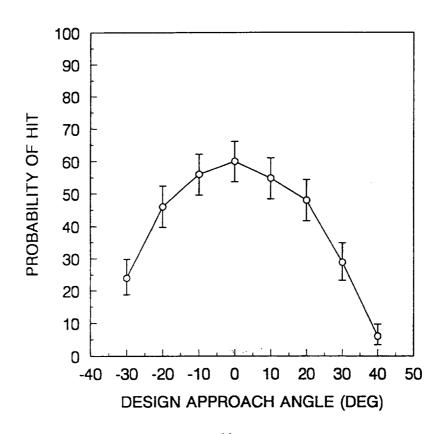
# MOE REQUIRES ACCURATE MODEL

- (1) STOCHASTIC
- (2) RADAR TRACK MODELING
- (3) AIRFRAME RESPONSE MODELING









#### SUMMARY AND CONCLUSIONS

MULTI-TIER METHOD: COMPLEX MOE

USEFUL FOR MULTIDISCIPLINARY WORK

COMBINE AERODYNAMIC, EW, LETHALITY, etc.

**EXAMPLE**:

SIMPLE MOE DOES NOT OPTIMIZE TRUE MOE

BEST MIDCOURSE A FUNCTION OF HOMING

### Using OTIS For Advanced Conceptual Design

An Example Of A Complex System Modeling And Trajectory/Performance Optimization Problem

Presented by:
Leon McKinney
McKinney Associates West
Huntington Beach, CA

To

The AIAA Atmospheric Flight Mechanics Conference Baltimore, MD, August 7, 1995

McKinney Associates West

#### An Overview of McKinney Associates West

- McKinney Associates West was formed in October, 1993, by Leon McKinney, after 11 years with McDonnell Douglas Aerospace as a senior systems performance analyst on space and defense projects
- Based in Huntington Beach, California, MAW offers system synthesis, modeling, analysis, and optimization solutions to problems in the aerospace & defense and commercial industries
- MAW also markets the capabilities of McKinney Associates and D. G. Purdy Associates for environmental and regulatory projects

# Recent Related MAW Projects

- Air-launched ATBM performance trade study, May 1994
- Beyond Visual Range (BVR) air-to-air missile performance trade study, April - September 1995

McKinney Associates West

McKinney Associates West

, A

For further information:

Leon McKinney

214 Knoxville Avenue

Huntington Beach, CA 92648

Phone: (714) 536-0289

Fax: (714) 536-2317

E-mail: leonmck@maw.com

#### **Using OTIS For Advanced Design Studies**

McKinney Associates West	
menniney Associates West	

In the fall of 1991, McDonnell Douglas Aerospace began an effort to assess for NASA-JSC air-launched space transportation concepts.

The author, then employed at McDonnell Douglas, was the principal analyst for the study, reporting to Dan Young and Bob Dawson.

After time-consuming efforts to model the air-launch concept using other trajectory modeling and optimization software, the study team decided to "take the plunge" and use OTIS - this was the first in-depth effort by the author to use OTIS on a real study, with real study timeline requirements.

This study turned out to be a prime example of the versatility and power of OTIS.

Air-Launched Space Transportation System Concept Analysis

#### **Rationale for Air-Launch Systems**

McKinney Associates West	

- Catastrophic Shuttle failure during space station assembly or ops will shut down the manned space program for an extended period
- An alternate manned transportation system is neccessary to ensure manned access to space
- Safety is THE MOST IMPORTANT issue for future manned space transportation systems
- Air-launch with a single stage is a good approach:
  - Launch at higher altitudes is inherently safeer lower dynamic pressures, altitude buys time to abort
  - Minimum parts
  - In-line mating of spacecraft to booster provides greater reliability
  - Non-powered glide landing (like Shuttle) is familiar

#### **Air-Launch Study Objectives And Goals**

McKinney Associates West

#### **Objective:**

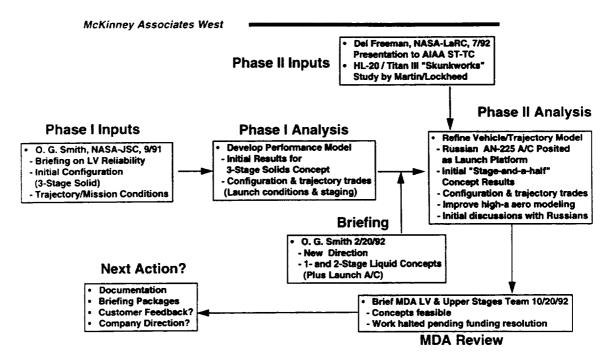
• To examine the feasibility of air-launched manned space transportation systems

#### Goals:

- Provide an optional apporach to the Shuttle for delivering both manned and unmanned payloads to LEO
- Significantly reduce reliability and safty risk over current Shuttle and ELV launch systems
- Significantly reduce costs of manned space transportations

Air-Launched Space Transportation System Concept Analysis

#### Air-Launch Study Approach & History



#### **Study Groundrules & Assumptions**

mortime,	y maaddialea west				
<ul><li>Spacecraft</li></ul>	Modeled by N	NASA/LaRC HL	-20; Injection	Wt approx.	25.000 lbs

Aircraft Russian AN-225; payload capability = 551,000 lbs

•Launch Vehicle Single or multiple stages; total stage weight NTE 525,000 lbs

Propellents LOX/LH2

Engines Igntion T/W > 1.0, Isp > 431 sec vaccuum

•Launch Altitude = 45,000 ft, Mach = 0.9, Flight path angle = 0 degrees

Minimum of 2 seconds after aircraft/launch vehicle separation •Ignition Delay

1100 psf, 32 degrees •Max Q/Alpha

McKinney Associator Woot

 Target Orbit 220 nmi circular, inclination = 28.5 degrees

Spacecraft abort capability required during aircraft and launch Abort

vehicle flight

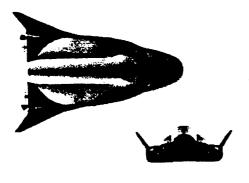
Air-Launched Space Transportation System Concept Analysis

#### **Launch Platform Aircraft Comparision**

McKinney Associates West Take-off **Payload** Weight Weight 747-200F 775,000 lb 247,800 lb C5B 837,000 lb 261,000 lb ANTONOV AN-225 1,322.750 lb 551,000 lb

#### Lockheed/Martin "Skunk Works" HL-20 Configuration

McKinney Associates West

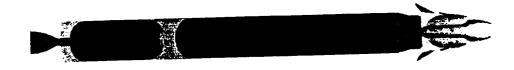


HL-20 Subtotal Dry Weight (w/ 22% Margin)	19,170 lb
Personnel & Gear	1,953
HL-20 "Useful Load"	21,123
HL-20 Propellants & Fluids	4,363
HL-20 "Wet Weight" (Injection Weight)	25,486
Launch Escape System (LES) Propellent	3,685
Launch Vehicle Adapter and LES Motors (Dry)	4,629
Total PLS System Weight Above LV	33,800 lb

Ref: HL-20 PLS "Business-As-Usual" vs. "Skunkworks" Comparison Study Results Presentation to A. Aldrich, NASA HQ, 6/25/92

Air-Launched Space Transportation System Concept Analysis

#### "Stage-and-a-Half" Air-Launch Configuration



- Launch platform Russian AN-225 (Mriya "Dream")
- Single propulsion stage w/ one SSMEclass engine,at 100% rated thrust
- Vehicle ascends to Space Station orbit apogee of 220 nmi. Then two options examined: (1) stage performs circularization burn; (2) PLS performs circularization burn.

500,000 lbs	
45,000 ft	
871 ft/sec	
0.90	
0.00 degree	\$
2.00 sec	
512,300 lbf	
497,190 lbf	
0.994	
453.5 sec	
381.4 sec	
	45,000 ft 871 ft/sec 0.90 0.00 degree 2.00 sec 512,300 lbf 497,190 lbf 0.994 453.5 sec

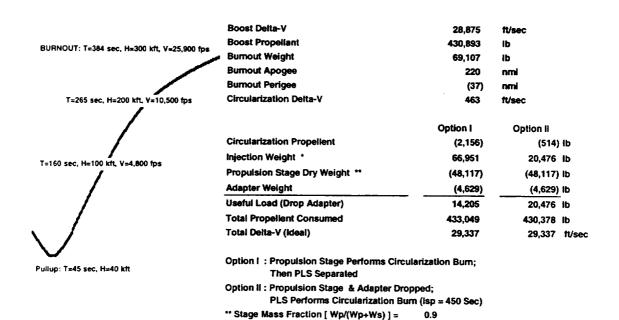
#### **Combined Configuration**

McKinney Associates West



Air-Launched Space Transportation System Concept Analysis

#### "Stage-and-a-Half" Performance Summary



#### "Stage-and-a-Half" Detailed Performance

McKinney Assoc	ciates West				Propulsion Stag	se Mass Fractio	n Sensitivity
Initial Weight	500,000	lbs			Option I	Option II	,
Initial Altitude	45,000	ft			76,420	76.040	0.85
Initial Velocity	871.268	ft/sec		Structure	70,496	70,145	0.86
Initial Mach	0.900				64,708	64,386	0.87
nitial Flight Path Angle (FPA)	0.00	degrees			59,052	58,758	0.88
gnition Delay Time	2.00	sec			53,523	53,256	0.89
(Alpha = 0)					48,117	47,877	0.90
Ingine Thrust (Vaccuum)	512,300	1bf			42,829	42,616	0.91
1 SSME-class engine @ 100% rated	thrust)				37,656	37,469	0.92
ngine Thrust (@ Alt)	497,190	Ibf			37,030	0.,	
gnition Thrust/Weight	0.994			A 1	-14.099	-11,198	0.85
Engine Vaccuum Isp	453.50	Sec		Payload	-8,174	-5.489	0.86
Weight Flow	1,129.66	1b/sec		(Injected PLS)	-2,387	89	0.87
Total Burn Time	381.44	sec		(Adapter Dropped)	3,270	5,540	0.88
Boost Delta-V	29,337	ft/sec			8.799	10.869	0.89
Boost Propellant	430,893	1b			14.205	16,079	0.90
Burnout Weight	69,107	No.			19,493	21,175	0.91
Burnout Apogee	220.00	nmi			24,665	26,160	0.92
Burnout Perigee	-37.43	nmi			24,000	20,.00	*
Circularization Delta-V	462.52	ft/sec					
	Option I	Option II				e Mass Fraction	For
and the state of t	2,156	514	lb		Injected PLS V	Veight≖	21,123 lt
Circularization Propellent	66,951	20,476	lb				
Injection Weight	48,117	48,117	lb		Option 1	Option II	
Propulsion Stage Dry Weight *	4,629	4,629	lb		0.913	0.910	
Adapter Weight	14,205	20,476	lb				
Useful Load (Drop Adapter)	433,049	431,407	ib				
Total Propellent Consumed Total Delta-V (Ideal)	29,800	29,800	ft/sec				

Option 1: Propulsion Stage Performs Circularization Burn; Then Adapter Dropped

Option II : Propulsion Stage & Adapter Dropped; PLS Performs Circularization Burn (Isp = 450 Sec)

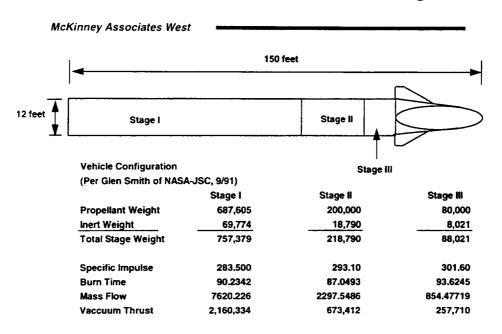
\* Stage Mass Fraction [ Wp/(Wp+Ws) ] = 0.90

Air-Launched Space Transportation System Concept Analysis

#### What Makes Air-Launch Systems Attractive

- Performance advantage of air-launch vs. ground-launch IS NOT due solely to the "free" velocity imparted by the launch platform, nor is it the extra potential energy derived from launching at 40-45 kft
- The real reason for air-launch's performance gain is significantly reduced gravity and drag losses
- Air-launching at high altitude and zero flight path angle reduces gravity losses
- Ground-launch requires vertical climb-out of the dense lower atmosphere quickly, to reduce drag losses and aerothermal loads; however in the process gravity losses are incurred.
- Air-launch is above most of the atmosphere no vertical climb-out required
- Launching at high altitudes reduces drag and aerothermal loads, allowing higher angles of attack; thus enabling more optimal control
- This reduction of losses enables air-launch concepts to add the required energy for orbit more efficiently than conventional ground-launch concepts.

#### Air-Launch vs. Ground-Launch Trade Configuration



Air-Launched Space Transportation System Concept Analysis

#### Air-Launch = Lower Losses + Better Efficiency

Components of Mission Delta-V Required (ft/sec)	Ground Launch	Air Launch	Air Launch Im	pact
Orbital Velocity	25,146	25,146	0	0%
Total Trajectory Losses	4,915	3,434	(1,481)	-30%
Gravity Loss	4,249	2,295	(1,954)	-46%
Drag Loss	222	526	304	137%
Steering / Thrust-Vector Loss	150	574	423	282%
Back Pressure Loss	309	64	(245)	-79%
Additional/Misc. Loss/(Gain)	(15)	(25)	(10)	67%
Initial Velocity (Earth Rotation)	(1,340)	(1,340)	0	0%
Initial Velocity (Kinetic Energy From Launch A/C)	o	(871)	(871)	
Total Mission Delta-V Required	28,721	26,369	(2,352)	-8%

#### Air-Launch vs. Ground-Launch Performance Summary

	Ground-Launch 3- Stage Solids	Air-Launch 3- Stage Solids	Air-Launch in	npact
Fixed Payload Useful Load Injected (lbs)	21,123	21,123	0	0%
Delta-V Required (fps)	28,721	26,369	(2,352)	-8%
Gross Liftoff Weight (GLOW) (lbs)	974,298	638,688	(335,610)	-34%
Fixed Stages Useful Load Injected (Ibs)	25,022	40,309	15,286	61%
Delta-V Required (fps)	28,721	26,369	(2,352)	-8%
Gross Liftoff Weight (GLOW) (lbs)	1,101,889	1,113,491	11,601	1%

#### Notes:

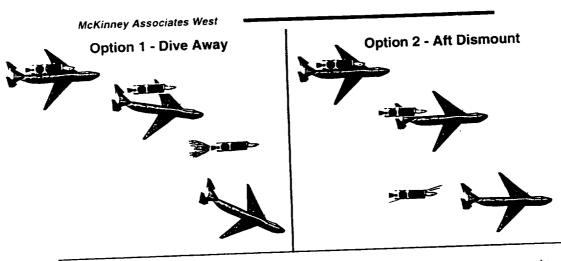
- (1) NASA-JSC Baseline Stage Propellent and Inert Weights
- (2) Stage Impulsive Velocities (Delta-Vs), Mass Fractions, and Isp all held fixed; stage size varied to produce Baseline Injected PLS Weight (see (3) below)
- (3) Baseline Injected PLS ("Useful Load") defined as "Skunk Works" PLS subtotal dry weight, plus personnel & gear :

19,170 lbs 1,953 lbs 21,123 lbs

- (4a) For Ground-Launch, the NLS adapter/launch escape system and the LES propellent are subtracted from the injected Weight to arrive at the Useful Load
- (4b) For Air-Launch, no LES propellent assumed

Air-Launched Space Transportation System Concept Analysis

#### **Representative Air-Launch Options**



- Separation is an important issue. First-look analysis indicates that sufficient clearance can be
  generated. Option 1 assumes (1) zero-lift flight (a = 0) for the launch vehicle during ignition delay and (2)
  the AN-225 pulls a 2g dive, Option 2 assumes drag-chute extraction of the launch vehicle, via rails on
  top of the AN-225.
- Additional analyses are required: discussions with DAC or other authorities (i.e. Russians) on AN-225;
   btter definition of high-alpha aerodynamics; examinations of structural attachments for HL-20 to launch vehicles (NASA-LaRC); and operational requirements issues (TBD)

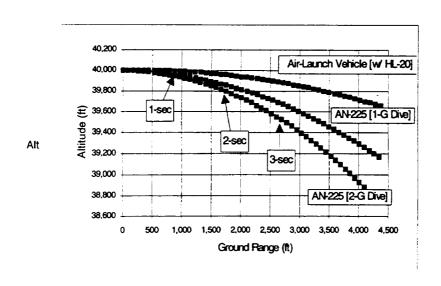
#### **Technical Issues Requiring Further Investigation**

McKinney Associates West

- (1) Separation is the the most important issue: "how we get the launch vehicle off the back of the AN-225 (or other launch platform)?" First-look analysis indicates that, assuming (1) zero-lift flight (alpha = 0) for the launch vehicle during ignition delay and (2) launch platform pulling a 2g dive, sufficient clearance can be generated (see backup chart). Other options include drag-chute extraction of the launch vehicle, via rails either on top of the launch platform, or from the cargo hold. Need discussions with NASA on Shuttle drop tests and other authorities (i.e. Russians) on AN-225
- (2) Better definition of high-alpha aerodynamics
- (3) Examinations of structural attachments for HL-20 to launch vehicles (NASA-LaRC)
- (4) Operational requirements issues (TBD)

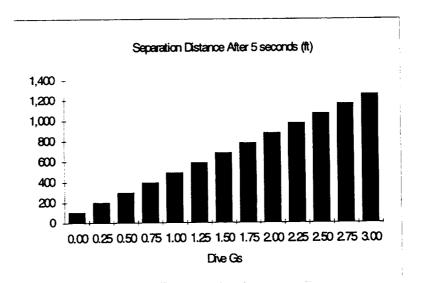
Air-Launched Space Transportation System Concept Analysis

#### **Separation Analysis Results**



#### Separation Analysis: Distance = f(Dive Gs)

McKinney Associates West



Air-Launched Space Transportation System Concept Analysis

#### **Air-Launch Study Summary**

McKinney Associates West

#### **CONCLUSIONS:**

- Air-launch HL-20 is feasible no major technical obstacles identified
- Air-launch HL-20 is a viable option to Shuttle and Shuttle derivatives
- The Russian AN-225 is the only airplane available with sufficient payload capability
- Concept has growth potential for both manned and unmanned missions
- Concept competitive with Spacelifter and SSTO concepts (for missions with over 20 klbs to LEO

#### **MAJOR ISSUES:**

- Availability and suitability of Russian AN-225 aircraft
- Safe launch vehicle / aircraft separation and abort modes
- Lauch vehicle stability and control during initial launch sequence
- Engine availability for launch vehicle
- Mass fraction of launch vehicle stage
- Recurring costs of launch vehicle expendable stage and engine

#### **Recommendations From MDA Study**

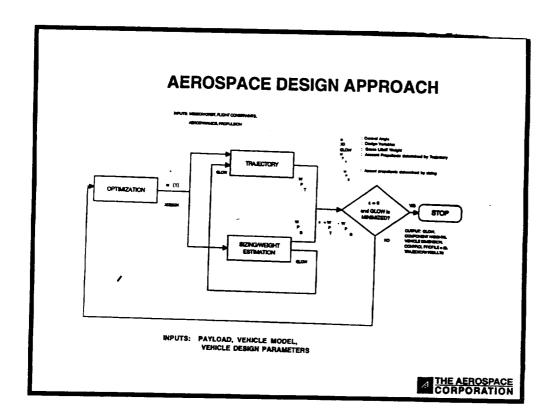
McKinney Associates West	
merchancy recording trest	

- Obtain more detailed AN-225 aircraft information from Russian sources and study aircraft to launch vehicle interfaces
- Obtain more detailed HL-20 configuration/interface information
- Resolve critical technical issues (LV/aircraft separation, LV stability and control) and resign concept design and performance
- Develope preliminary program information (schedule/costs/ricks estimates)
- Study potential applications of air-launch systems
- Study reusability potential of air-launch systems

#### APPLICATION OF FONSIZE TO BI-PROP AND TRI-PROP SSTO DESIGNS

HAI N. NGUYEN SPACE LAUNCH OPERATIONS

THE AEROSPACE CORPORATION



#### **GROUND RULES AND ASSUMPTIONS**

- SELECT THE TRI-PROPELLANT OPTION
- USE NASA SIZING/WEIGHT EQUATIONS (EACH EQUATION REPRESENTS A SUB-SYSTEM) TO SIZE THE VEHICLE FOR THE FOLLOWING REQUIREMENTS
  - 45 KLB DUE EAST (BURN OUT IN 50X100 NMI ORBIT)
  - 25 KLB TO SSF ORBIT (51.6 DEG, 220 NMI ORBIT)
- USE TWO ADVANCED CRYOGENIC OMS ENGINES TO PERFORM C'RCULARIZATION MANEUVERS
- USE EIGHT RD-701 ENGINES WITH THROTTLE SETTING DETERMINED DURING THE OPTIMIZATION TO PROVIDE 1.2 T/W AT LIFT-OFF
- USE RASV AERODYNAMICS WITH APPROPRIATE SCALING TO REFLECT NASA DESIGN



#### **GROUND RULES AND ASSUMPTIONS**

FLIGHT CONSTRAINTS

FLIGHT CONSTRAINT PARAMETERS	DESIGN VALUE NOT TO BE EXCEEDED
- MAXIMUM DYNAMIC PRESSURE - MAXIMUM Q-ALPHA	850 PSF +/-1600 PSF-DEG
- MAXIMUM AXIAL LOAD	3 G

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#### **GROUND RULES AND ASSUMPTIONS**

#### • DESIGN CONSTRAINTS

CONSTRAINT PARAMETERS	DESIGN VALUE
- WING LOADING	54.6 PSF
- MODE 1 KEROSINE PROPELLANT FRACTION	0.1502
- MODE 1 LH2 PROPELLANT FRACTION	0.0581
- MODE 1 LOX PROPELLANT FRACTION	0.7917
- TOTAL KEROSINE PROPELLANT FRACTION	0.0983
- TOTAL LH2 PROPELLANT FRACTION	0.0782
- TOTAL LOX PROPELLANT FRACTION	0.8235
- LIFT-OFF T/W	1.2



#### **RD-701 ENGINE DATA**

	MODE 1	MODE 2
VAC THRUST (Ib)	2x449,618	2x176,475
VAC ISP (sec)	415.06	462.06
FLOW RATES (lb/sec)	2x856.42	2x327.44
_H2	2x65.05	2x54.46
KEROSINE	2x162.51	N/A
EXPANSION RATIO EXTENSION DOWN	170	170
EXTENSION UP	70	N/A
CHAMBER PRESSURE (psi)	4350	2175

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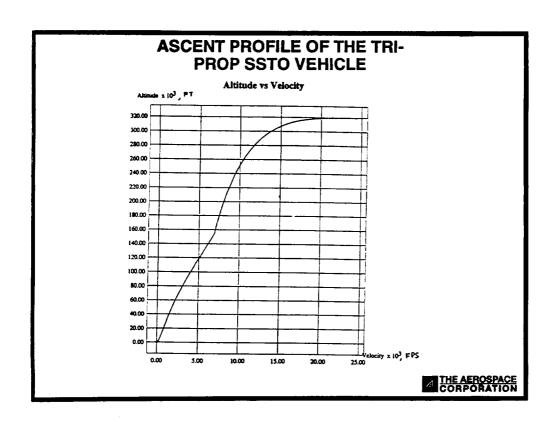
#### **ANALYSIS**

- VALIDATE NASA SSTO PERFORMANCE AND SIZE
- ASSESS SSTO DESIGN SENSITIVITY:
  - TANK UNIT WEIGHTS
  - TPS UNIT WEIGHTS
  - STRUCTURAL UNIT WEIGHTS
  - ULTIMATE SAFETY AND NORMAL LOAD FACTORS IN WING DESIGN
  - REDUCTION FACTORS
  - WING LOADING
  - ENGINE T/W
  - ENGINE ISP



VALIDATION OF PERFORMANCE AND SIZE

THE AEROSPACE CORPORATION



#### **COMPARISONS OF TRI-PROPELLANT SSTO DESIGN WEIGHTS**

1.0	WING EXPOSED WING SURFACE CARRY-THROUGH	AEROSPAG WEIGHT ( 10940. 1746.	_	NAS WEIGH 9273. 1542.		<u>∆%</u> +17.3
2.0	TAIL	1740.	2257.	1342.	1899.	+18.85
	·· <del>-</del>				1000.	*10.00
3.0	BODY LHZ TANK KEROSENE LOZ TANK BASIC STRUCTURE SECONDARY STRUCTURE	15805. 2786. 12605. 19149. 12197.	62542.	15779. 2778. 12577. 19021. 12194.	62349.	+0.3
4.0	INDUCED ENVIRONMENT PROTECTION TPS FUSELAGE WING INTERNAL INSULATION PURGE, VENT, DRN, & HAZRD GAS DET	18729. 13143. 5586. 969. 720.	29418.	17892. 13123. 4769. 968. 713.	29573.	+4.32
5.0	UNDERCARRIAGE AND AUX. SYSTEMS NOSE GEAR MAIN GEAR	1142. 6557.	7 <del>699</del> ,	1040. 5976.	7016.	+9.7
6.0	PROPULSION, MAIN ENGINES PRESS AND FEED HELIUM PNEUMATIC & PURGE SYSTEM GEMBALS ENGINE MOUNTED HEAT SHIELD	38062. 12203. 2428. 2672. 1411.	<b>56</b> 776.	40735. 9795. 2389.	52919.	7.29
7.0	PROPULSION, REACTION CONTROL (RCS) THRUSTERS AND SUPPORTS PROPELLANT TANKS DISTRIBUTION & RECIRCULATION VALVES	507. 1398. 1309. 569.	3783.	507. 1240. 1309. 569.	3626.	4.33
8.0	PROPULSION, ORBITAL MANEUVER (OMS) ENGINES PROPELLANT TANKS PRESSURIZATION	545. 831. 1113.	2490.	545. 740. 990.	2275.	9.45

#### COMPARISONS OF TRI-PROPELLANT SSTO DESIGN WEIGHTS

		AEROSPACE WEIGHT (LB)	NASA WEIGHT (LB)	<u>Δ%</u>
25.0	RESERVE FLUIDS	E202.	7289.	
26.0	INFLIGHT LOSSES	3806.	3804.	
27.0	PROPELLANT, MAIN START-UP LH2 KEROSENE LIO2 ASCENT LH2 KEROSENE LO2	2148923. 32641. 1958. 4113. 26570. 2116282 165493. 20031. 1742758.	2143459. 32121. 1927. 4047. 26147. 2111338. 165206. 207443. 1738689.	0.255
28.0	PROPELLANT, REACTION CONTROL ORBITAL PROPELLANT ENTRY PROPELLANT	3249. 2463. 786.	2886. 2192. 695.	
29.0	PROPELLANT, ORBITAL MANEUVER	21768.	193 <del>69</del> .	4.22
GROS	790-T3LJ 23	2421574.	2382988.	1.619

#### COMPARISONS OF TRI-PROPELLANT SSTO DESIGN WEIGHTS

		AEROSPACE WEIGHT (LB)	NASA WEIGHT (LB)	Δ%
9.0	PRIME POWER FUEL CELL SYSTEM BATTERIES	2339. 2324. 15.	2339. 2324. 15.	0
10.0	ELECTRIC CONVERSION AND DISTR. POWER CONVERSION CIRCUITRY ELECTROMECH. ACT. (EMA) CABLING EMA CONTROL UNITS	6374. 1705. 4199. 113. 356.	6331. 1705. 4199. 103. 324.	0.68
11.0	HYDRAULIC CONVERSION AND DISTR.	0.	0.	0
12.0	CONTROL SURFACE ACTUATION	1413.	1285.	9.96
13.0	AVIONICS	1314.	1314.	0
14.0	ENVIRONMENTAL CONTROL	2396.	2395.	0.04
15.0	PERSONNEL PROVISIONS	0.	0.	0
18.0	PAYLOAD PROVISIONS	0.	0.	0
19.0	MARGIN (15%)	27373	2612 L	
	EMPTY	209859.	200257.	4.79
20.0	PERSONNEL.	0.	0.	0
21.0	PAYLOAD ACCOMMODATIONS	0.	0.	0
22.0	PAYLOAD	45000.	45000.	0
23.0	RESIDUAL AND UNUSABLE FLUIDS	13408.	13044.	2.79

#### ASSESSMENT OF DESIGN SENSITIVITY

THE AEROSPACE CORPORATION

#### ASSESSMENT OF TRI-PROPELLANT SSTO DESIGN SENSITIVITY

	DESIGN VALUE		DESIGN	SENSITIVITY.
TANKS				
- LH <sub>2</sub> UNIT WEIGHT - LOX UNIT WEIGHT	0.364 0.458	LB/FT <sup>3</sup> LB/FT <sup>3</sup>	140,518. 91,939.	LB/LB/FT <sup>3</sup> LB/LB/FT <sup>3</sup>
STRUCTURE BODY				
- NOSE - INTERTANK - AFT - THRUST STRUCTURE	1.11 1.64 4.0 0.0021	LB/FT <sup>2</sup> LB/FT <sup>2</sup> LB/FT <sup>2</sup> _B/LB	1,514. 14,638. 3,308. 12,059,256.	LB/LB/FT <sup>2</sup> LB/LB/FT <sup>2</sup> LB/LB/FT <sup>2</sup> LB/LB/LB
PROPULSION				
- REDUCTION FACTOR - ENGIN T/W - MODE 1 I <sub>SP</sub> - MODE 2 I <sub>SP</sub>	0.25 70.2 415.06 462.06	SEC SEC	-184,743. -1,974. -822. -1,215.	LB/UNIT LB/UNIT LB/SEC LB/SEC
TPS				
WING UNIT WEIGHT     WING REDUCTION FACTOR     FUSELAGE UNIT WEIGHT     FUSELAGE REDUCTION FACTOR	1.287 0.268 1.152 0.268	LB/FT <sup>2</sup>	15,764. -27,715. 41,532. -65,362.	LB/LB/FT <sup>2</sup> LB/UNIT LB/LB/FT <sup>2</sup> LB/UNIT
WING & CARRY THROUGH				
- ULTIMATE SAFETY	1.75	G	18,603.	LB/G
LOAD FACTOR - NORMAL LOAD FACTOR	2.0	G	19,614.	LB/G
WING REDUCTION FACTOR	0.4		-66,264.	LB/UNIT

<sup>•</sup> CHANGE IN DRY WEIGHT DUE TO VARIATION IN DESIGN VALUE

#### ASSESSMENT OF TRI-PROPELLANT SSTO DESIGN SENSITIVITY

	DESIGN VALUE		DESIGN SENSITIVIT		
CARRY THROUGH REDUCTION FACTOR	0.4		-10,591.	LBAUNIT	
WING LOADING	54.6	LB/FT <sup>2</sup>	-1,281.	LBALB/FT <sup>2</sup>	
PAYLOAD	45.	KLB	2.1655	LIMLB	
MARGIN	0.15		577,537.	LB/UNIT	

#### WEIGHT IMPACTS DUE TO UNATTAINABLE UNIT WEIGHT (OR COEFFICIENTS) GOALS (TRI-PROP DESIGN) WEIGHT **ATTAINED IMPACT** PROPOSED TANKS 0.4655 lb/ft3 14,263. **1**b - LH2 0.364 lb/ft3 18.020. In 0.654 lb/ft3 0.458 lb/ft3 - LOX 32,283. lb STRUCTURE (15% OFF) 0.0024 1b/1b 3,618. lb 0.0021 lb/lb - THRUST 252. lb 1.2765 lb/ft2 1.11 16/ft2 - NOSE 1.886 lb/ft2 3,601. Ib 1.64 lb/ft2 - INTER TANK 1.985. Ib 4.0 lb/ft2 4.6 lb/ft2 · AFT 9,456. lb TPS (15% OFF) 3,044. Ib 1.287 lb/ft<sup>2</sup> 1.4801 lb/ft2 - WING 7.177. b 1.3248 lb/ft2 1.152 lb/ft<sup>2</sup> - FUSELAGE 10,221. **b**

SUBTOTAL:

51,960. **b** 

#### WEIGHT IMPACTS DUE TO UNATTAINABLE REDUCTION FACTOR GOALS (TRI-PROP DESIGN)

	PROPOSED	ATTAINED	WEIGHT IMPACT
WING .	40%	20%	13,253. lb
CARRY THROUGH	40%	20%	2.118. lb
	•		15,371. 16
TPS			
WING	26.8%	13.4%	3,714. lb
FUSELAGE	26.8%	13.4%	8.758. lb
,			12,472. lb
PROPULSION			
MAIN ENGINS	25%	12.5%	23,093. lb
		SUBTOTAL:	50,936. Ib

#### WEIGHT IMPACTS DUE TO OFF-NOMINAL OPERATIONS AND OPERABILITY CONSIDERATIONS (TRI-PROP DESIGN)

	PROPO	SED	ENHANG FOR OP		
MAIN PROPULSION					
- MODE I ISP	415.06	5	405.06	5	8,220. <b>lb</b>
- MODE 2 ISP	462.06	5	452.06	\$	12,150. lb
- T/W (15%)	70.2		59.67		20.786. lb
					41,156. lb
WING					
- WING LOADING	54.6	ıь∕n²	44.6	₽b/ft <sup>2</sup>	12,810. lb
- ULTIMATE SAFETY FACTOR	1.75	G	2.0	G	4,641. Ib
NORMAL LOAD FACTOR	2.0	G	2.25	G	4.903. lb
					22,364. lb
			SUBTOTAL		63.520. lb

	•	

## E arth S atellite P rogram

Diane D. Buell

7 August 1995

**MITRE** 

#### **Overview**

• Description: User-friendly coverage, tracking, and

mapping program

• Scope: Any elliptical Earth orbit

• Utility: System design trade-off analyses for both

ground and space segments

• Host: Macintosh II

• Language: THINK Pascal

**MITRE** 

#### **Major Features**

- Satellite ground tracks
- Elevation angles, ranges, range rates
- Outage zones
- Spot beam projections
- Integrated mapping capability (color with zoom option) as well as tabular output
- Standard Macintosh user interface
- Quick turnaround

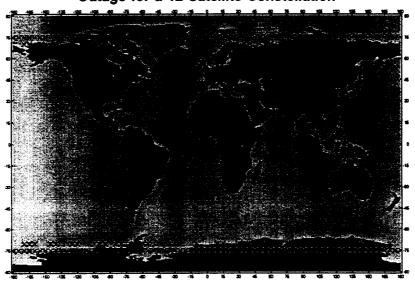
MITRE

Fri, Aug 4, 1995, 12:28 PM

MSC outage

Page 1

#### Outage for a 12 Satellite Constellation



- x zone 1: 0.0 to 3.0 hrs. + zone 2: 3.0 to 6.0 hrs. zone 3: 6.0 to 9.0 hrs. □ zone 4: 9.0 to 12.0 hrs. zone 5: 12.0 to 15.0 hrs.
- zone 6: more than 15.0 hrs.

#### Outage Inputs

#### Satellite Number 1

Semi-major axis Eccentricity a = 16732.0 km e = 0.0  $i = 55.0^{\circ} \text{ north}$   $\Omega = 0.0^{\circ} \text{ east}$   $\omega = 0.0^{\circ} \text{ east}$ Inclination
Longitude of the node

Doing note of the note  $\omega = 0.0^{\circ}$  east Time of perigee  $\omega = 0.0^{\circ}$  east Time of perigee  $\tau = \text{Jan 1, 1991 12:00:00 AM}$ . The time of the osculating values is Jan 1, 1991 12:00:00 AM.

Satellite mass N⁄Α Effective area for drag N/A
Coefficient of drag 2.0
Effective area for solar radiation
Absorption/reflectivity constant 1.5

#### Satellite Number 2

Semi-major axis a=16732.0 kmEccentricity e=0.0Inclination  $i=55.0^{\circ}$  north Longitude of the node Argument of perigee  $\omega=90.0^{\circ}$  east  $\tau=10.0^{\circ}$  Time of perigee  $\tau=10.0^{\circ}$  axis  $\tau=10.0^{\circ}$ 

Satellite mass N/A SateInte mass NVA
Effective area for drag N/A
Coefficient of drag 2.0
Effective area for solar radiation
Absorption/reflectivity constant 1.5

Fri, Aug 4, 1995, 12:54 PM

MSC outage

Page 7

Page 1

#### **Outage Inputs**

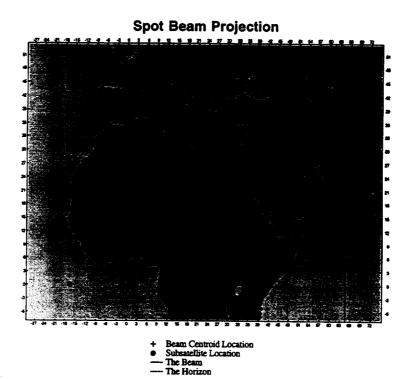
#### Inputs For This Run

Latitude of first site Latitude of last site Latitude step size -90.0° north 90.0° north 5.0° north -180.0° east

Longitude of first site Longitude of last site Longitude step size 180.0° east 5.0° east

Starting time of the run Ending time of the run Time step Jan 1, 1991 12:00:00 AM Jan 2, 1991 12:00:00 AM 0.1 hour

Minimum elevation angle Size of each outage zone 3. Minimum number of satellites 3.0 hours required for coverage



Fri, Aug 4, 1995, 12:53 PM

Kuwait 5.0 bw

Page 1

#### Spot Beam Inputs

# Satellite Information Semi-major axis a = 42163.0 kmEccentricity e = 0.0Inclination $i = 20.0^{\circ}$ north Longitude of the node $\Omega = 200.0^{\circ}$ east Argument of perigee $\tau = 200.0^{\circ}$ east Time of perigee $\tau = 200.0^{\circ}$ east The time of the osculating values is Apr 30, 1991 12:00:00 AM. Satellite mass Effective area for drag Coefficient of drag Coefficient of drag Effective area for solar radiation Absorption/reflectivity constant Starting time of the run Ending time of the run Ending time of the run Time step Contours of equal loss were not calculated. Number of beams a = 42163.0 kmE 42163.0 km E 42163.0 km E

Beam	Centroid
Number	Longitude
1	49.00°

Centroid Latitude 30.00° Beam Width 5.00°

#### **Mathematics**

Equations of motion

$$\dot{r} = -\frac{\mu}{r^3} r + \mathbf{F}$$

where

$$\mathbf{F} = \mathbf{f_{J}}_2 + \mathbf{f_{drag}} + \mathbf{f_{solar\ radiation}}$$

- Fourth-order Runge-Kutta algorithm propagates satellite orbits
- Initial conditions determined from the values of the six classical orbital elements known at some instant

MITRE

#### **Example Applications**

- Constellation design for air traffic control
- Beam coverage for Desert Storm
- Elevation angles for communications
- Satellite terminal evaluator design
- Constellation coverage assessment

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#### **Future Enhancements**

- Simplified interface for special orbit types and/or two-body option
- Additional calculations (e.g., azimuth angle and higher derivatives)
- Time history plots of parameters
- Cursor driven control
- Beam projections for constellations
- Spreadsheet and word processor compatibility

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#### REPORT DOCUMENTATION PAGE

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of trajectory optimization prog work. Users of trajectory optim	grams were invited to presen mization programs, particula vere also encouraged to prese que forum to exchange ideas	t a 15–20 minute summarly those supporting flig	ming and applications. The developers ary of recent improvements or current the programs, utilizing new approaches their applications and/or modifications. en industry, academia, and government			

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